ABSTRACT

SAINI, ADITYA. Leading-Edge Flow Sensing for Aerodynamic Parameter Estimation. (Under the direction of Dr. Ashok Gopalarathnam.)

The identification of inflow air data quantities such as airspeed, angle of attack, and local lift coefficient on various sections of a wing or rotor blade provides the capability for load monitoring, aerodynamic diagnostics, and control on devices ranging from air vehicles to wind turbines. Real-time measurement of aerodynamic parameters during flight provides the ability to enhance aircraft operating capabilities while preventing dangerous stall situations.

This thesis presents a novel Leading-Edge Flow Sensing (LEFS) algorithm for the determination of the air-data parameters using discrete surface pressures measured at a few ports in the vicinity of the leading edge of a wing or blade section. The approach approximates the leading-edge region of the airfoil as a parabola and uses pressure distribution from the exact potential-flow solution for the parabola to fit the pressures measured from the ports. Pressures sensed at five discrete locations near the leading edge of an airfoil are given as input to the algorithm to solve the model using a simple nonlinear regression. The algorithm directly computes the inflow velocity, the stagnation-point location, section angle of attack and lift coefficient. The performance of the algorithm is assessed using computational and experimental data in the literature for airfoils under different flow conditions. The results show good correlation between the actual and predicted aerodynamic quantities within the pre-stall regime, even for a rotating blade section.

Sensing the deviation of the aerodynamic behavior from the linear regime requires additional information on the location of flow separation on the airfoil surface. Bio-inspired artificial hair sensors were explored as a part of the current research for stall detection. The response of such artificial micro-structures can identify critical flow characteristics, which relate directly to the stall behavior. The response of the microfences was recorded via an optical microscope for flow over a flat plate at different freestream velocities in the NCSU subsonic wind tunnel. Experi-

ments were also conducted to characterize the directional sensitivity of the microstructures by creating flow reversal at the sensor location to assess the sensor response. The results show that the direction of microfence deflection correctly reflects the local flow behavior as the flow direction is reversed at the sensor location and the magnitude of deflection correlates qualitatively to an increase in the freestream velocity. The knowledge of the flow-separation location integrated with the LEFS algorithm allows the possibility of extending the LEFS analysis to post-stall flight regimes, which is explored in the current work.

Finally, the application of the LEFS algorithm to unsteady aerodynamics is investigated to identify the critical sequence of events associated with the formation of leading-edge vortices. Signatures of vortex formation on the airfoil surface can be captured in the surface-pressure measurements. Real-time knowledge of the unsteady flow phenomena holds significant potential for exploiting the enhanced-lift characteristics related to vortex formation and inhibiting the detrimental effects of dynamic stall in engineering applications such as helicopters, wind turbines, bio-inspired flight, and energy harvesting devices. Computational data was used to assess the capability of the LEFS outputs to identify the signatures associated with vortex formation, i.e. onset of vortex shedding, detachment, and termination. The results demonstrate useful correlation between the LEFS outputs and the LEV signatures.

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Leading-Edge Flow Sensing for Aerodynamic Parameter Estimation

by
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DEDICATION

To my parents and my beloved sister for their unwavering support and love over all these
years.

BIOGRAPHY

Aditya earned his Bachelors degree in Mechanical engineering in 2012 from the Indian Institute of Technology in Ropar, India. During his junior year, he received the opportunity to work as an intern in the Engine Test-Bed Research and Design center (ETBRDC) at Hindustan Aeronautics Limited. The experience of working in the R&D department greatly motivated him to pursue the path of research. In 2012, Aditya enrolled at North Carolina State University for a M.S degree in Aerospace Engineering. His research interest in aerodynamics led him to join the NCSU Applied Aerodynamics Research Group under the guidance of Dr. Ashok Gopalarathnam. Aditya completed his Masters degree in 2014 and continued for a Ph.D. His broad research interests involve applied aerodynamics, computational fluid dynamics, wind-tunnel testing, dynamics modeling and simulations.

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LIST OF SYMBOLS

A_0	leading-edge Fourier term coefficient
AoA	angle of attack, deg
c	airfoil chord
C_d	airfoil drag coefficient
C_l	airfoil lift coefficient
$C_{l,max}$	airfoil maximum lift coefficient
C_n	airfoil normal force coefficient
C_p	coefficient of pressure, $\frac{p - p_\infty}{q_\infty}$
$C_{P,peak}$	minimum pressure coefficient value
C_s	airfoil suction force coefficient
f	separation-point location as % of chord
f_c	airfoil camber function
f_t	airfoil thickness function
$FADS$	flush air-data sensing
$LEFS$	leading-edge flow sensing
$LESP$	leading-edge suction parameter
LEV	leading-edge vortex
LUT	look-up table
MAV	micro-aerial vehicle
NLF	natural-laminar-flow
P	local static pressure
P_o	total pressure
P_∞	freestream static pressure
q_∞	dynamic pressure, $\frac{1}{2}\rho U_\infty^2$
r	parabola leading-edge radius

Re	Reynolds number based on airfoil chord length
RMS	root mean square
t	airfoil maximum thickness
U_{∞}	freestream velocity
UAE	unsteady aerodynamics experiment
UAV	unmanned aerial vehicle
x_{stag}	x -component of stagnation-point location
x_{peak}	pressure-peak location
y_{stag}	y -component of stagnation-point location
α	angle of attack, deg
α_{0l}	zero-lift angle of attack
α_{stall}	airfoil stall angle of attack
γ	vorticity
ρ	freestream density
λ	inclination of parabola
ϕ	potential function
ψ	stream function

Chapter 1

Introduction

In-flight determination of the sectional aerodynamic state of a wing can be essential for load monitoring, stall sensing, and envelope protection. Aerodynamic state here refers to the magnitude of the incoming flow velocity, U_∞ , effective angle of attack, α , and the sectional force coefficients, C_l and C_d . Real-time determination of the operating state of various wing and tail sections could be used for the control of flying bodies in different flight scenarios.

It is believed that birds and insects exploit such sensing from various mechanoreceptors in their wing to fine tune their flight performance and control [1, 2]. For instance, birds can sense the slightest changes in the air pressure which enables them to stabilize in varying wind gusts and changing weather [3, 4]. Bats have the remarkable ability to reverse flight directions at high speeds in short distances [5]. These outstanding capabilities are possible due to the fact that bats and birds can interact effectively with the surrounding airflow by sensing crucial information such as the flow speed, stagnation point, stall, and turbulence over the wing through numerous mechanoreceptors [6]. For these reasons, pressure sensing using flush surface pressure ports on wing surfaces are being investigated for use on small air vehicles [7, 8, 9]. Extensive research is dedicated to developing sophisticated on-skin sensors for monitoring lifting surfaces [10], and potential control strategies that will use the sectional pressure and shear information for aircraft attitude control [11, 12].

Real time measurement of aerodynamic parameters during aircraft flight is critical for flight control augmentation. The ability to determine the air-data parameters in real time holds the key to accurately predict impending stall and for carefully guiding the aircraft throughout its flight envelope. Dangerous and unrecoverable stall situations like deep stall can be anticipated and successfully avoided by possessing accurate knowledge of the complex flow environment. Such stall-warning devices and algorithms mostly operate based on the information of local angle of attack and/or the stagnation-point location.

Traditional techniques used for the determination of airspeed and angle of attack are mainly multi-hole probes mounted on intrusive air data booms [13], electromechanical vanes [14], or flush air data sensing (FADS) systems using surface pressure measurements on the aircraft nose cone [15, 16, 17, 18, 19, 20]. A typical aircraft boom that measures pitot-static pressures and flow angles is shown in Figure 1.1 and Figure 1.2 shows the air data boom of F-35C combat aircraft.

Figure 1.1: A conventional airdata boom for measuring pressures and flow angles.[14]

Although undoubtedly successful, these techniques have certain drawbacks. For instance, the installation of these devices may disturb the flow, or in certain scenarios the probes are constrained to specific locations on the aircraft and are too bulky for application on small unmanned vehicles. Air data booms also suffer from errors due to misalignment and vibration. Although used extensively during flight tests of prototype aircraft as well as on routine commercial operations, these approaches provide only the global aerodynamic state (α, β and velocity) of the aircraft. In FADS technology, pressure measurements in these investigations are acquired

Figure 1.2: Air data boom of F-35C combat aircraft. Image source: http://commons.wikimedia.org/wiki/File:Cockpit_and_Air_Data_Boom_F-35C.jpg.

using a matrix of surface pressure ports located on the aircraft nose or on the wing sections in the vicinity of the leading edge. Figure 1.3 shows the nose cap of the F-18 research aircraft. In most of the research efforts, the pressure ports are typically accommodated in the vicinity of the fuselage nose-tip because the flow in this region remains unseparated over a greater angle-of-attack regime. But this approach has drawbacks associated with complex design of the avionics housed in the aircraft nose. An approach that provides aerodynamic states of multiple sections of the lifting surfaces will provide information on the spanwise distribution of the aerodynamic loading. A variation of the FADS system is an approach in which the pressure measurements on the leading-edge of a wing are used to determine the aerodynamic state of the section [21]. However, in all the implementations of the FADS approach described in literature, either extensive experimental calibration is required via flight tests or wind tunnel campaigns, or powerful mathematical tools are employed.

A majority of these mathematical schemes are based on artificial neural networks [22, 23, 24], which offer an approach to model complex non-linear systems without the need of explicit knowledge about the functional correlation that exists between the input and output variables of the system. However, they require substantial amounts of training data to develop the relationship

Figure 1.3: F-18 Systems Research Aircraft closeup of nose cap showing new flush air data system sensor holes from <http://www.dfrc.nasa.gov/Gallery/Photo/F-18SRA/HTML/EC97-43936-6.html>.

between the input pressure values and the required aerodynamic parameters. Also, they do not capture the actual physics of the flow and instead act as mathematically modeled "black-boxes".

Additionally, inertial navigation systems [25] and GPS information is extensively used to navigate and control small UAVs and MAVs, but these methods focus on measuring the rigid-body dynamics and ignore the actual aerodynamic flow conditions on the wings of the UAVs. Vision-based systems [26, 27, 28] are also gaining recent attention for controlling the flight of autonomous vehicles because of their low weight and capability of detecting the surrounding environment. However, there is high computation cost associated with the video/image processing algorithms and the vehicle is still susceptible to unpredictable wind gust and weather conditions.

The approaches mentioned above lack the crucial aerodynamic information that can be helpful in mimicking the extraordinary capabilities possessed by birds for extremely maneuverable

and agile flight. Knowledge of the section operating condition can be used to effectively adapt wing geometry using cruise flap [29, 30] or other geometry changes. Flow-field sensing on the horizontal tail from a few surface pressure measurements can be used to detect the loss of effectiveness of the tail when they are affected by wake impingement and allow real-time estimates of the control surface performance. Apart from routine aircraft operations, the knowledge of the aerodynamic operating condition could be useful in wind tunnel or flight tests of prototype configurations for deducing the aerodynamic causes behind flight behaviors.

The potential benefits of knowing the aerodynamic state extend to several non-aerospace applications as well. For example, on horizontal axis wind turbines, in which peak power generation requires operation of the blades close to aerodynamic stall, the uncertainties in time-varying wind speed and direction, induced inflow, planetary boundary layer, and structural flexibility of the blade make it difficult to deduce the instantaneous aerodynamic operating condition of blade sections. On race car wings and yacht sails and keels, unquantifiable and time-varying factors influence the state of the aerodynamic surfaces, making an approach that deduces the aerodynamic state from a few pressure measurements valuable.

1.1 Research Objective

The aerodynamic flow phenomenon over wings can largely be analyzed using pressure based flow-field estimation techniques over the surface of the airfoil. For instance, the pressure distribution along the chord of an airfoil can be integrated to calculate the sectional loads and moments on lifting surfaces. The pressure gradients can provide vital indication of separation and transition locations. The maximum pressure location in the vicinity of the leading edge corresponds to the location of the stagnation point. It has been shown in earlier works [31] that the location of the stagnation point varies monotonically with the airfoil angle of attack and lift coefficient.

Hence, the surface pressure information can be effectively used to develop successful flight control schemes for aircrafts and UAVs [32]. A major drawback of this approach is that the

accuracy of deducing flow field information highly depends on the number of the surface pressure ports used. For instance, if the freestream static pressure, p , and fluid density, ρ , are available, the stagnation pressure (approximated as maximum pressure location from pressure ports) can be used to deduce the velocity magnitude from the following equation:

$$P_0 = P + \frac{1}{2}\rho U^2 \quad (1.1)$$

If the leading edge is instrumented with densely-packed pressure ports, then the maximum value of the measured pressures will serve as a good approximation to p_0 , yielding both s_{stag} and p_0 , from which α and U can be easily deduced. However, practical considerations limit the number of pressure ports that can be placed near the leading edge. With a few pressure ports, it is unlikely that the stagnation point location will coincide with a port location, hence resulting in erroneous deduction of velocity. To illustrate the error from such a process, experimental data from wind tunnel tests performed at the Ohio State University of the S809 airfoil [33] is used. In these experiments, the airfoil model was instrumented with pressure taps and the freestream velocity was set to different known values (corresponding to test-section flow speeds for desired Reynolds numbers). For this illustration, five pressure taps on the model are selected as representative of sparsely-distributed ports on a flush air data system at the leading edge. These port locations correspond to one at the leading edge ($x/c = 0$), two on the upper surface at $x/c = 1.4\%$ and 4.5% , and two on the lower surface at $x/c = 1.5\%$ and 5.6% . For a range of angles of attack and freestream velocities, the maximum of the pressures measured at these five ports for each case was used as the estimated p_0 , from which U was estimated. This estimated U was compared with the known U from the wind tunnel tests. Figure 1.4 compares the estimated and actual freestream velocities. The comparison illustrates how the error in U varies with α depending on whether or not the stagnation point happens to coincide with a port location. It is seen that the deduced value of U can have an error as high as 17.48%.

In essence, reducing the number of surface pressure sensors can greatly reduce the accuracy of the predicted aerodynamic state parameters, and using a large number of pressure ports at

Figure 1.4: Estimated freestream velocities (U) using the maximum of the pressure measured at five ports for the S809 airfoil compared with the actual velocities (dashed lines).

different wing sections is not always feasible in practical applications. The ability to derive the sectional aerodynamic parameters accurately with a few sparsely distributed pressure ports is the main focus of this research.

The current work derives inspiration from the benefits of FADS systems and introduces a novel method for extracting the aerodynamic quantities from minimum number of pressure measurements near the leading edge of the airfoil. The Leading-Edge Flow Sensing (LEFS) algorithm makes use of the pressure distribution from an exact solution for inviscid flow past an infinite parabola. Most airfoils have leading-edge shapes that closely resemble a parabola. The pressures measured at the few pressure ports are used to best fit the exact solution for flow over a parabola to determine the leading-edge pressure distribution, and thus estimate the required sectional aerodynamic characteristics.

1.2 Thesis Outline

The second chapter discusses the inviscid equations of flow past a parabola that form the foundation of the complete aerodynamic model used in the algorithm along with the associated parameters and approximations. In Chapter 3, the methodology adopted in testing the algorithm is discussed along with the non-linear regression method is used for solving the aerodynamic model. The chapter also covers the procedure followed for fitting a parabolic curve over the leading edge of any airfoil (cambered or symmetric). It also explains the significance of the coefficients that appear in the equations and the process involved in extracting the required air data parameters from these coefficients. The method is tested for different airfoils using pressure data obtained from computations and from wind-tunnel tests. The results for all the test cases are presented in Chapter 4. Chapter 5 discusses the techniques available for the detection of separation-point location as an indicator of aerodynamic stall and presents the results for a micropillar-based approach explored in this research. The conclusions based on the results and the future scope of this research are discussed in Chapter 6.

Chapter 2

Background

Flows near the leading edges of airfoils have been of particular interest to understand the physical behavior associated with complex aerodynamic flows like boundary-layer separation and dynamic vortex formation. Many researchers have taken advantage of simplified flow solutions to comprehend the characteristics of such complex flow phenomena. In this research, the equations of flow past a parabola are used as an approximation for the flow near the leading edge of an airfoil. This chapter discusses the reasoning behind such an approximation. The inception of this research idea is highlighted in Sec. 2.1 and the analytical solution for flow past a parabola is derived using potential flow theory and conformal mapping in Section 2.2. Finally, Sec. 2.3 explains the aerodynamic model used in the LEFS algorithm.

2.1 Flow near the Leading Edge of an Airfoil

Most airfoils designed for subsonic flow have a non-zero leading-edge radius. The upper and lower surfaces of an airfoil can be generally described by a polynomial of the form:

$$y = \pm T_o x^{1/2} + C_1 x \pm T_1 x^{3/2} + C_2 x^2 + .. \quad (2.1)$$

where T_n and C_n are coefficients for the thickness and camber respectively, and x is the distance

measured from the leading edge to the trailing edge of the airfoil. The first coefficient, T_o , determines the nose radius of the airfoil (r) and the second coefficient, C_1 , corresponds to the initial slope of the camberline of the airfoil (λ). In a small region in the proximity of the leading edge where $x = O(t^2)$ and t is the thickness ratio of the airfoil, the airfoil can be expressed by the first two terms of the Eq. 2.1 using r and λ as follows:

$$y = \pm\sqrt{2rx} + \lambda x \quad (2.2)$$

Equation 2.2 defines the profile of an inclined parabola. Hence, it is an acceptable assumption to approximate the airfoil nose with the equation of a parabola.

The earliest analysis of approximating the airfoil profile with a trivial shape in the vicinity of its edges was directed at correcting the leading-edge singularity that arises in potential flow about a round-nosed thin airfoil. Thin airfoil theory breaks down in the vicinity of the leading edge of an airfoil. The potential solution for flow past a parabola was used by researchers (Van Dyke [34], Lighthill [35], Jones [36]) to develop mathematical rules or techniques that would render the solution for flow around an airfoil uniformly valid near the edges. The ratio of the exact solution for the simple shape to its formal thin-airfoil theory expansion served as a multiplicative factor that corrects the formal second-order solution for the actual airfoil near the edge. This technique was further extended to higher approximations, compressible flow, three-dimensional wings, sharp edges, and slender bodies of revolution [37, 38, 39].

Later, the development of numerical schemes to solve Navier-Stokes equations made it possible to study the flow past semi-infinite bodies to shed light on comprehensive local solutions at corners, vertices, edges, and other regions of high interest. Again, the flow past a parabola gained attention because of the motivation to study the phenomenon of boundary-layer separation at the leading edge. Theoretical studies of separation at the nose of an airfoil were presented by Werle and Davis [40, 41], Botta et al. [42], Veldman [43], Stewartson [44], Dennis and Walsh [45, 46], Ruban [47], and Rusak [48]. The laminar boundary layer equations were

solved for incompressible flow past a parabola at a single angle of attack. Flow separation from the surface of the airfoil has significant effects on the aerodynamic performance. The ability to model the influence of adverse pressure gradients and skin friction using numerical solutions on the surface of a parabola at a non-zero angle of attack made such studies ideal for detailed examination of boundary layer separation, with the additional advantage of reduced computational effort compared to flow past a full airfoil.

Further, more complicated flows due to unsteady aerodynamics have also been investigated for a parabola [49, 50] to gain insight into the actual flow behavior over the leading edges of conventional airfoils. Such studies were aimed at understanding the fundamental mechanisms governing the unsteady effects associated with vortex formation, growth and detachment, when an airfoil undergoes pitching or oscillatory motion. Dynamic stall was simulated by varying the stagnation-point location on the parabola to mimic the unsteady motion of the airfoil leading edge with respect to the free stream. Bhaskaran et al. [50] also showed that the unsteady flows past leading edges of thin airfoils can be considered in isolation without a significant loss in accuracy by comparing the results with full-airfoil simulations.

Most of the previous research efforts that treated the airfoil leading edge as a parabola have been directed towards developing theoretical models to understand the complex flow physics associated with leading-edge flows. The current research employs the same fundamental approach, but is geared towards flow-sensing applications. This work shows that the measurement of a minimum number of pressures at the leading edge of the airfoil can be used in conjunction with a model developed for flow past parabola to deduce the aerodynamic quantities in real time.

2.2 Potential Theory and Conformal Mapping

Potential flows are ‘ideal’ flows which are defined as irrotational and inviscid. The fluid in a potential flow follows the contours of the solid surface, and no boundary layer is formed. These assumptions make it possible to derive the equations for basic elementary flows mathematically

in terms of the stream function, ψ , or the velocity potential, ϕ . The stream function relates to the velocity components in such a way that the continuity is satisfied and the scalar velocity potential emerges from the irrotational flow condition, which ensures that the vorticity (or the curl of velocity, $\nabla \times V$) is zero at every point. The velocity potential is complementary to the stream function, and both must satisfy the Laplace equation ($\nabla^2 F = 0$). Since the Laplace equation is linear, various flow solutions can be added to obtain other complex flow cases. Superposition of these elementary flows in different ways can be used to derive more complicated flows making potential flow theory a valuable tool for developing low-order aerodynamic models.

The elementary flow solutions from potential theory are successful in representing complex physical flows but in certain scenarios the geometric structure makes the formulation inconvenient. In such cases, conformal mapping is used as an effective tool to appropriately map the flow along a complex geometry to a much simpler physical domain. Conformal mapping is an important mathematical technique used in complex analysis and is used to extend the application of potential flow theory to practical aerodynamic flows. Grid points are mapped between two different analytic domains in the complex plane. A mapping between complex planes is basically an equation that transforms the field of points defined on a domain in the ζ -plane, $\zeta = \xi + i\eta$, to the z -plane, $z = x + iy$, according to a mapping function, $z = g(\zeta)$. Figure 2.1 illustrates such a transformation.

An important property of such conformal maps is that the grid lines orthogonal in the ζ -plane will remain orthogonal in the z -plane after the transformation. In essence, a conformal map preserves the angles locally. If the function is harmonic (i.e. it satisfies Laplace's equation), then the transformation of such functions via conformal mapping is also harmonic. This property allows equations pertaining to any field that can be represented by a potential function (all conservative fields) to be solved via conformal mapping.

The equations of flow past a parabola are derived by using the complex potential function for a stagnation-type flow and then transforming that equation using conformal mapping to obtain the solution for flow past a parabola as described in the following subsections.

Figure 2.1: Grid map transformation from ζ -plane to z -plane using $z = g(\zeta) = \frac{1}{2}(\zeta^2 + 1)$.

2.2.1 Stagnation-Point Flow

The potential-flow equations that mimic stagnation-point behavior can be represented in the complex plane, where $\zeta = \xi + i\eta$. The complex potential is given by the following equation:

$$F(\zeta) = \frac{1}{2}(\zeta + \beta - i)^2 \quad (2.3)$$

where the real and imaginary part represent the velocity potential and stream function respectively.

The derivative of this complex potential in the ζ -plane gives the complex velocity, which becomes:

$$F'(\zeta) = \frac{dF}{d\zeta} = (\zeta + \beta - i) \quad (2.4)$$

Figure 2.2 shows the streamlines for the stagnation-point flow.

Figure 2.2: Stagnation-point flow.

2.2.2 Flow Past a Parabola

The stagnation-point flow can be mapped onto the physical z -plane, using the transformation:

$$z = \frac{1}{2}(\zeta^2 + 1) \quad (2.5)$$

The complex velocity in the z -plane can be obtained by applying the transformation from Eq. 2.5 to the complex velocity in the ζ -plane:

$$F'(z) = \frac{dF}{dz} = \frac{dF}{d\zeta} \frac{d\zeta}{dz} \quad (2.6)$$

$$F'(z) = \frac{\zeta + \beta - i}{\zeta} \quad (2.7)$$

$$F'(z) = 1 + \frac{\beta - i}{\zeta} \quad (2.8)$$

$$F'(z) = 1 + \frac{\beta - i}{\sqrt{2z - 1}} \quad (2.9)$$

The real and imaginary part of the complex velocity in the z -plane are connected to the velocity components in the x and y direction, respectively, given as $u = \text{Re}\{f(z)\}$ and $v = -\text{Im}\{f(z)\}$. On the surface of the parabola, the velocity can be written in terms of a coordinate

Figure 2.3: Flow past a parabola.

system in which the parabola is written as $Y = \pm\sqrt{2X}$, where X and Y are non-dimensional coordinates with respect to the radius of the parabola, r . That is, $X = x/r$ and $Y = y/r$. Figure 2.3 shows the stagnation-point flow transformed into the flow past a parabola. Rewriting the complex velocity in the $X - Y$ coordinates gives:

$$F'(Z) = 1 + \frac{\beta - i}{\sqrt{2(X + iY) - 1}} \quad (2.10)$$

$$F'(Z) = 1 + \frac{\beta - i}{\sqrt{2(\frac{Y^2}{2} + iY) - 1}} \quad (2.11)$$

$$F'(Z) = 1 + \frac{\beta - i}{\sqrt{Y^2 + 2iY - 1}} \quad (2.12)$$

$$F'(Z) = 1 + \frac{\beta - i}{\sqrt{(Y + i)^2}} \quad (2.13)$$

$$F'(Z) = 1 + \frac{\beta - i}{(Y + i)} \quad (2.14)$$

Further algebraic manipulation to separate the the real and imaginary terms gives:

$$F'(Z) = \frac{Y(Y + \beta)}{Y^2 + 1} - i \frac{Y + \beta}{Y^2 + 1} \quad (2.15)$$

Therefore, the velocity components on the surface of the parabola are:

$$u_x = \frac{Y(Y + \beta)}{Y^2 + 1} \quad (2.16)$$

$$u_y = \frac{Y + \beta}{Y^2 + 1} \quad (2.17)$$

Substituting the dimensional coordinates back in the equation using $X = x/r$ and $Y = y/r$, the net velocity on the surface of the parabola can be written as:

$$U = \sqrt{u_x^2 + u_y^2} \quad (2.18)$$

$$U = \frac{y + r\beta}{\sqrt{y^2 + r^2}} \quad (2.19)$$

Further transforming the equation in terms of the x-coordinate using $y = \pm\sqrt{2rx}$ gives:

$$U = \frac{\pm\sqrt{2rx} + r\beta}{\sqrt{2rx} + r^2} \quad (2.20)$$

$$U = \frac{\sqrt{x} \pm \frac{\sqrt{r}\beta}{\sqrt{2}}}{\sqrt{x + \frac{r}{2}}} \quad (2.21)$$

For the freestream velocity U_∞ , this expression is equal to the equation given by van Dyke as :

$$\frac{U}{U_\infty} = \frac{\sqrt{x} \pm a'}{\sqrt{x + \frac{r}{2}}} \quad (2.22)$$

The positive sign and negative sign refer to the upper and lower surface, respectively. Substituting $a' = \beta\sqrt{r/2}$, The final expression used in this research for the velocity on the surface of a parabola near the leading edge is:

$$\frac{U}{U_\infty} = \frac{A(\sqrt{x} \pm a')}{\sqrt{x + \frac{r}{2}}} \quad (2.23)$$

where A is a multiplicative factor, of order one, which relates the flow on the parabola to that on the actual airfoil. Since the flow solution is used to fit real pressure measurements, this

parameter also compensates for viscous effects during the nonlinear regression process discussed in Chapter 3.

2.3 Aerodynamic Model

The aerodynamic model used in the LEFS algorithm is built on the equations of flow past a parabola as derived in the previous section. The final expression for the surface velocity shown in Equation 2.23 can be converted to the coefficient of pressure. Using Bernoulli's equation, the difference between the surface pressure and the known freestream static pressure can be written as:

$$P - P_{\infty} = \frac{1}{2} \rho U_{\infty}^2 \left[1 - \left(\frac{U}{U_{\infty}} \right)^2 \right] \quad (2.24)$$

On combining Eqs. 2.23 and 2.24, we obtain:

$$P - P_{\infty} = \frac{1}{2} \rho U_{\infty}^2 \left(1 - A^2 \frac{(\sqrt{x} \pm a')^2}{x + r/2} \right) \quad (2.25)$$

Figure 2.4: Parabola at an angle of attack.

In Eq. 2.25, the freestream static pressure, P_∞ , and density ρ are assumed to be known quantities, and the parabola nose radius, r , and slope, λ , are the known geometry parameters based on the parabola-airfoil matching, which is explained in Chapter 3. Eq. 2.25 is expressed for a symmetric parabola in the parabola frame of reference as depicted in Fig. 2.4. Hence, for the case of a cambered airfoil, the coordinates in this equation are transformed to the airfoil frame of reference based on the inclination of the defined parabola that appropriately fits the airfoil.

Figure 2.5: A comparison of the pressure distribution on a cambered S809 airfoil obtained from XFOIL with the pressure distribution over a parabola obtained from Eq. 2.25. Here, r and λ in the equations are estimated by generating the parabolic curve based on the leading-edge region of S809 airfoil, as discussed in Chapter 3.

To illustrate the application of this analytical model, the pressure distribution over a parabola calculated using the above equation is compared with the pressures over an airfoil obtained using the XFOIL code in Fig. 2.5. The flow behavior over a parabola accurately mimics that over an airfoil in a region in proximity to the stagnation-point location, which lies on the lower surface for the $\alpha = 5^\circ$ and on the upper surface for $\alpha = -5^\circ$ on the upper surface. Hence, the pressure distribution over the parabola can be inversely used to solve for the variables U_∞ , a' , and A by curve fitting it with the actual pressures measured over the airfoil under

consideration. Five pressures were deemed sufficient to successfully perform this operation and estimate the relevant flow characteristics. The following chapter illustrates the methodology adopted for the application of this approach in greater detail.

Chapter 3

Methodology

This chapter presents the methodology adopted to deduce the sectional aerodynamic parameters from the pressures measured near the leading edge of the airfoil. The equation for flow past a parabola, discussed in Chapter 2, serves as the fundamental aerodynamic model used in the LEFS technique. In addition, there are critical steps involved in the LEFS algorithm that are integral to successfully solving the aerodynamic model. The approach adopted for fitting the parabola to the airfoil leading-edge is presented in Section 3.1 and the nonlinear regression method is discussed in Section 3.3. Finally, the process of converting the LEFS output into relevant aerodynamic quantities is explained in Sec. 3.4.

3.1 Parabola-Airfoil Matching

The LEFS algorithm exploits the parabolic nature of the airfoil geometry in the proximity of the leading edge. The airfoil leading edge radius, r , and the initial slope of the airfoil camberline, λ , are the parameters required to generate the parabolic curve that fits the leading-edge region of the airfoil, according to van Dyke [34]. However, the flow solution in Equation 2.25 based on this parabola is valid only in extreme proximity to the leading edge of the airfoil. In practical applications, such limited space is insufficient for the installation of sufficient pressure taps/sensors in that region. Therefore, a different approach is adopted in which a parabolic profile is generated

in the initial step using a curve-fit approach, that matches the airfoil leading-edge coordinates with the equation of a parabola given in Equation 3.1 until 5% of the airfoil chord. Consequently, new values of the parameters r and λ are determined based on this parabola-airfoil matching that make the solution for flow past a parabola (based on new r and λ) acceptable till about 5% of the chord.

$$y = \pm\sqrt{2rx} + \lambda x \quad (3.1)$$

3.1.1 Symmetric Airfoil

For a symmetric airfoil, the quantity λ (in Equation 3.1) drops out and the equation is reduced to an elementary form, $y = \pm\sqrt{2rx}$. This also implies that, at $\alpha = 0$ the stagnation point coincides with the leading edge on the airfoil and the parabola. The equation of a symmetric parabola is solved with the known airfoil coordinates in the leading-edge region (until 5% of the chord) via a curve-fitting approach, to estimate a new leading-edge radius. The new value of r is used in the aerodynamic model in Eq. 2.25 to render the solution acceptable for a considerable parabolic envelope on the airfoil nose. Figure 3.1 shows the parabolic profile generated using the new r for a symmetric NACA 0012 airfoil.

3.1.2 Cambered Airfoil

For a cambered airfoil, an ideal angle of attack is defined as the angle of attack at which the stagnation point coincides with the leading edge of the airfoil. To define a parabolic profile that models the nose of the airfoil, it is crucial to ensure that the stagnation-point location on the leading-edge of the airfoil closely matches with that on the constructed parabola. The α_{ideal} for the airfoil is estimated using inviscid velocity distributions obtained from the airfoil design and analysis code, XFOIL[51]. The slope of α_{ideal} determines the new inclination, λ , for the parabola. The leading-edge radius of the parabola that fits the airfoil nose reasonably well is estimated by following a trivial curve-fit approach as adopted for the case of a symmetric airfoil. The steps involved are listed as follows:

(a) Leading-edge region.

(b) Full airfoil.

Figure 3.1: Leading-edge parabola fit for a symmetric NACA 0012 airfoil.

1. Start with an initial guess value of r and generate a symmetric parabola.
2. Rotate this parabola by $\theta = \alpha_{ideal}$ about a point on the camberline of the airfoil with the x -coordinate equal to r .
3. Compare the coordinates of the rotated parabola with the airfoil nose-region till 5% of the chord by evaluating a cumulative error function, $\delta = \sum_{i=1}^N (y_i^p - y_i^a)^2$ (superscripts p and a represent the parabola and airfoil respectively).
4. Determine the value of r iteratively till the δ is minimized.

Figure 3.2 compares the parabola-fit for a cambered airfoil. The new value of r determined here is used directly in Eq. 2.25. The aerodynamic model in LEFS is solved in a frame of reference attached to the axis of the parabola. Therefore, for a cambered airfoil the pressure tap locations on the airfoil are mapped into the frame of reference of the rotated parabolic profile used to fit

(a) Leading-edge region.

(b) Full airfoil.

Figure 3.2: Leading-edge parabola fit for a cambered S809 airfoil.

the airfoil nose.

3.2 Selection of pressure-port locations

The aerodynamic model presented in Section 2.3 solves for three output parameters, U_∞ , A , and a . A notable approximation in this simplified model is that it is derived from the inviscid equations of flow past a parabola. As a result, the direct input of all the five measured pressure values in the algorithm for any aerodynamic orientation might not yield accurate results, which is evident by observing the pressure distribution over the parabola shown in Figure A.7 in Chapter 2. The prime reason for utilizing five pressure data points is to be able to envelope the leading-edge region and select pressure data points near the stagnation-point location. At the stagnation point, the streamline is incident normally on the airfoil surface and divides the

flow on the upper surface from that on the lower surface. Since the boundary-layer formation commences from this point onwards, the boundary layer is relatively thin in this region, thereby exhibiting close resemblance to the inviscid pressures in the vicinity of the stagnation point. The pressure difference between the first pressure ports aft of the leading edge on the upper and the lower surface provides a reliable indication of whether the airfoil is operating at a positive or a negative angle of attack. Based on that information the pressure-tap information corresponding to the location of the stagnation point is used as depicted in the flow chart shown in Figure 3.3.

3.3 Nonlinear Regression

The aerodynamic model, for known P_∞ , r and λ , is provided in Equation 2.25. The aerodynamic model can be represented by F , which is a function of the location of the i^{th} pressure port (x_i), a' , A , and U_∞ . If the measured pressure at the i^{th} pressure port is P_i , then the error, ϵ_i , is defined as the difference between the pressure measured by the sensor and the pressure estimated from the function F , given as:

$$\epsilon_i = P_i - F(x_i, a', U_\infty, A) \quad (3.2)$$

The sum of the errors for all n pressure ports under consideration is:

$$S = \sum_{i=1}^n \epsilon_i^2 \quad (3.3)$$

The algorithm minimizes this residual function S to solve for the quantities a' and U_∞ . Numerous optimization techniques are available that use iterative procedures to solve such regression problems. The current procedure uses the nonlinear least-squares function (*lsqnonlin*) from the optimization toolbox in MATLAB to estimate the aerodynamic variables. This function uses the trust-region-reflective method for the non-linear curve fitting operation, which is based on the interior-reflective Newton method described in Ref. [52].

Figure 3.3: Flowchart.

3.3.1 Trust-Region Reflective Method

Iterative optimization techniques can be broadly studied under two categories: line-search methods and trust-region methods. Line-search methods calculate a search direction in each iteration

and then advance along this path to solve for a better solution, which is utilized in the following iteration to identify the new search direction. Trust-region algorithms construct an approximate model near the current best point and then solve the model to yield a better solution. The process is repeated until the final best solution is achieved. However, the approximate models are only valid in a “trust” region in the vicinity of the current iterate. Generally, a merit function is devised to track the performance of the model in the trust region. Consequently, it can limit or enlarge the size of the trust region. A review of different trust-region algorithms is presented by Yuan. [53]. The approach used in the *lsqnonlin* function in MATLAB is developed by forming a model with an appropriate quadratic function and scaling matrix. In addition to greater reliability and robustness, these methods have strong convergence properties and the capability to effectively handle bound constraints [54].

3.4 Determining Aerodynamic Parameters

The methodology presented in the previous sections provides a step-by-step description of the LEFS technique. The steps are summarized with the aid of a flowchart shown in Figure 3.3. Five pressure values sensed on the airfoil surface in the neighborhood of the leading-edge are used as inputs to the LEFS algorithm to return U , a , and A by employing nonlinear regression techniques. The local freestream dynamic pressure or velocity is obtained directly from the current approach. The analytical function derived for velocity distribution using the parabola model (Eq. 2.23) is differentiable and can be conveniently analyzed to obtain the local minimum and maximum values. The minimum value of the function in terms of a relates to the stagnation-point location on the airfoil surface as:

$$x_{stag} = a'^2 \tag{3.4}$$

$$y_{stag} = -a' \sqrt{2r} \tag{3.5}$$

Similarly, the pressure-peak location can be estimated from the maximum value of the velocity distribution as:

$$x_{peak} = \frac{r^2}{4a'^2} \quad (3.6)$$

substituting this value in the C_p distribution over a parabola results in:

$$C_{p_{peak}} = 1 - A^2 \frac{(r + 2a'^2)}{r} \quad (3.7)$$

In addition to capturing the pressure behavior in the leading-edge region of the airfoil, the parameters A and a' can be successfully used to extract additional aerodynamic information, such as the local angle of attack and lift coefficient. Two approaches were explored in this research to relate the LEFS output parameters to the local flow-field characteristics. The first approach utilizes asymptotic matching to directly relate a' and A to α , while the second approach relies on the stored look-up tables previously-developed using low-order simulation or CFD data. A detailed explanation of the two approaches is presented in the following subsections.

3.4.1 Direct Approach

Asymptotic matching is employed to derive a direct relationship between a , A and the local angle of attack. The process exploits the fact that the flow around an airfoil involves two disparate lengths, which means that the flow-field around the airfoil can be divided into two regimes based on the length scales. The inner region is defined near the nose of the airfoil, $x \sim O(r)$, where the flow is characterized by extreme velocity changes due to the presence of the stagnation-point and the suction peak and the length scale is governed by the leading-edge radius, r . The outer region is defined around the rest of the airfoil where the flow is dominated by classical thin airfoil theory. For viscous flows, the boundary layer also plays a significant role in the interaction between the two regions because the boundary layer formation initiates at the stagnation point in the inner region and develops into the outer region till the trailing

edge. However, in this analysis only the inviscid flow-field is matched in the near-nose region and the rest of the airfoil. Asymptotic theory is used to match the solutions in an overlapping domain. The existence of such an intermediate overlapping region implies that the inner limit of the outer region should agree with the outer limit of the inner region, to appropriate orders of magnitude [55].

The formulation for matching the inner and outer flow regions has been presented by researchers while investigating the problem of leading-edge stall of an airfoil. Rusak and Morris [56] performed the matching which accounted for the interactions between the near-wall viscous flow and the outer inviscid flow. Their analysis resulted in a model (simplified) problem of a uniform, compressible, steady stream past a semi-infinite parabola with a far-field circulation governed by a parameter that is related by the asymptotic matching to the angle of attack, nose radius of curvature, the airfoil camber and the flow Mach number. Tuck [57] had developed a similar matching approach but used the exact potential flow around the nose of the airfoil (replaced by a parabola) and matched in the far field with the thin-airfoil solution resulting in an expression relating the parameter β to the angle of attack, nose radius and the airfoil chord. This parameter is equivalent to the parameter β used in deriving the equation of flow past a parabola from stagnation-point flow in Section 2.2.2. In terms of the notation used in the current research, $\beta = a' A \sqrt{2/r}$.

In the current formulation, Tuck's approach for matching the solution on the parabola and the airfoil is followed. The outer matching provides a solution for the thin-airfoil theory in the inner limits, $x \rightarrow 0$, while the inner matching provides a result for non-dimensional parabola solution in the outer limits, $X \rightarrow \infty$. Similar-order terms from the limiting values of the inner and outer regions are matched to determine a relation between α , a' and A .

Outer Matching

The upper and the lower surface of an airfoil can be written in terms of the camber function, f_c , and the thickness function, f_t , as:

$$y = f_c(x) \pm f_t(x), \quad 0 \leq x \leq c, \quad (3.8)$$

For a small angle of attack, the rotated airfoil profile with respect to the freestream velocity takes the form:

$$y = -\alpha x + f_c(x) \pm f_t(x), \quad 0 \leq x \leq c, \quad (3.9)$$

The fluid velocity is $\nabla(Ux + \phi)$, where ϕ is the disturbance velocity potential due to the airfoil. It follows that the solution of Laplace's equation subject to the relevant boundary conditions, and the Kutta condition at the trailing edge, results in the airfoil streamwise disturbance velocity as:

$$u = \phi_x = \pm U\alpha \left(\frac{c-x}{x}\right)^{\frac{1}{2}} \pm \frac{U}{\pi} \left(\frac{c-x}{x}\right)^{\frac{1}{2}} \int_0^c \frac{f'_c(\xi)}{x-\xi} \left(\frac{\xi}{c-\xi}\right)^{\frac{1}{2}} d\xi + \frac{U}{\pi} \int_0^c \frac{f'_t(\xi)}{x-\xi} d\xi \quad (3.10)$$

The behavior of this thin airfoil solution near the leading edge can be simplified to:

$$u_x \rightarrow \pm U\alpha \left(\frac{c}{x}\right)^{\frac{1}{2}} \quad (3.11)$$

Inner Matching

Using the complex velocity for the parabola as derived in Section 2.2.2, the X -wise disturbance velocity can be obtained. The non-dimensional complex velocity for a parabola flow is:

$$F'(Z) = 1 + \frac{\beta - i}{(Y + i)} \quad (3.12)$$

For $|Z| \rightarrow \infty$,

$$F'(Z) \rightarrow 1 + \frac{\beta - i}{Y} \quad (3.13)$$

The X -wise disturbance velocity is the difference between the freestream velocity and the velocity on the parabola as $|Z| \rightarrow \infty$. This result is equal to $\pm\beta(2X)^{-\frac{1}{2}}$. This disturbance velocity is derived for the non-dimensional potential $F(Z)$. However, for the parabola problem, the actual complex potential is $UAF(z)$. The disturbance velocity on the parabola u_p becomes :

$$u_x \rightarrow \pm UA \frac{\beta\sqrt{r}}{\sqrt{2x}} \quad (3.14)$$

where $\beta = \frac{\sqrt{2}a'}{\sqrt{r}}$. Hence, the expression in terms of a' becomes:

$$u_x \rightarrow \pm UA \frac{a'}{\sqrt{x}} \quad (3.15)$$

Comparing the terms in Eq. 3.11 and Eq. 3.15 gives a relationship between the angle of attack and the parabola parameters, a' and A , as :

$$\alpha = \frac{a'A}{\sqrt{c}} \quad (3.16)$$

The matching produces an expression to estimate the angle of attack directly from the output parameters of the LEFS algorithm, i.e. a' and A . Further this result can be used with the thin-airfoil theory expression for the airfoil lift coefficient, which is:

$$C_l = 2\pi (\alpha - \alpha_{ol}) \quad (3.17)$$

$$C_l = 2\pi \left(\frac{a'A}{\sqrt{c}} - \alpha_{ol} \right) \quad (3.18)$$

where α_{ol} is defined as a zero-lift angle of attack and it can be estimated for any airfoil using

thin-airfoil theory. Furthermore, suction-force near the leading edge can also be obtained from the LEFS parameters. Leading-edge suction is caused by the stagnation point moving away from the leading edge to some other location when the airfoil is at an angle of attack. The flow stops at the stagnation point and must travel around the airfoil's leading edge towards the other surface. The importance of this force coefficient is better understood from the perspective of unsteady aerodynamics. The derivation for the suction-force coefficient, C_s , is presented in detail in Sec. 4.3.1. The expression relating C_s to a' and A takes the form:

$$C_s = 2\pi \left(\frac{a'A}{\sqrt{c}} - \frac{1}{\pi} \int_0^\pi \frac{df_c}{dx}(\theta) d\theta \right)^2 \quad (3.19)$$

where f_c is the camberline of the airfoil and θ is a transformation variable related to x as:

$$\theta = \cos^{-1} \left(1 - \frac{2x}{c} \right) \quad (3.20)$$

The inner-outer matching used in the current formulation is applicable solely to thin airfoil with thickness less than 15% of the chord. The inaccuracy for thick airfoils caused due to this limitation is demonstrated in Chapter 4, where the direct approach is assessed for thicker airfoil. Hence, a look-up table approach is proposed for extending the application of the LEFS algorithm to airfoils where thickness effects are more pronounced.

3.4.2 Look-up Table Approach

Due to the complexity associated with accommodating the thickness effects in the direct approach, a look-up table approach is adopted in the LEFS algorithm to estimate α and C_l from a' . The parameter a' , obtained from solving the aerodynamic model, strongly relates to the angle of attack and can be effectively used to generate the look-up tables. XFOIL is used to compute pressure data for several Reynolds numbers for a range of angles of attack encompassing both positive and negative stall angles. Five pressure values, extracted from the data, are processed using the LEFS algorithm to solve for a' . Two separate 2-dimensional look-tables

are generated with one having the α vs. a' values and the other having the α vs. C_l data. For any airfoil under consideration, such tables can be generated from computational data and integrated into the algorithm prior to the application. In real-time operation, the flow velocity solved from the aerodynamic model at any instant is translated to the local Reynolds number, which is then used along with a' to extract the angle of attack and lift coefficient from the tables. The effectiveness of the LEFS technique is evaluated under different flow conditions, and the results are presented in Chapter 4.

Chapter 4

Applications and Results

This chapter presents the application of the LEFS approach to different flow scenarios to characterize the performance of the LEFS technique. Computational simulations and experimental test data were used for various airfoils exhibiting different aerodynamic behavior. Section 4.1 shows the results of the LEFS algorithm applied to steady flow conditions. The capability of the technique for the identification of airfoil stall and the estimation of aerodynamic parameters beyond stall is explored in Sec. 4.2. Unsteady aerodynamic flows are investigated in Sec. 4.3 with the objective of capturing the signatures associated with the formation of leading-edge vortices using the outputs of the LEFS approach. Finally, the ability of the current approach to deduce the local sectional aerodynamic quantities for different sections of a rotating blade is tested and the results are presented in Sec. 4.4.

In all the investigations, five pressure measurements and their locations are extracted from the computational or experimental test data and this information is processed through the LEFS algorithm to determine the sectional characteristics like angle of attack, lift-coefficient, and freestream velocity, and the results are compared with the corresponding actual values available at the given flow conditions.

4.1 Steady Flow (Pre-Stall)

The LEFS algorithm was evaluated for steady flow conditions. Computational data available from previous research studies in the NCSU Applied Aerodynamics Research Lab was utilized for this study. Experimental data was gathered from National Renewable Energy Laboratory (NREL) wind-tunnel test campaigns conducted at the Ohio State University wind tunnel.

4.1.1 CFD Test Cases

NCSU’s REACTMB-INS code was used for CFD calculations and the details of the computations and validation studies are discussed in Ref. [58]. Steady flow calculations over the airfoil surface were performed for a range of angles of attack and the surface pressure data was post-processed to extract the pressure information at five discrete locations near the leading-edge. The five pressure values along with the respective locations on the surface are given as input to the LEFS algorithm, and the aerodynamic parameters are determined as described in Section 3.4. The test cases were evaluated for different airfoils: NACA 0012, SD 7003, and NACA 23012 at a Reynolds number of 3 million. The locations at which the pressures were extracted from the CFD data for all the three airfoils are tabulated in Table 4.1.

Table 4.1: Pressure port locations used to extract data for LEFS inputs from CFD data.

Airfoil	Pressure locations (x/c)
NACA 0012 (12% thick)	Lower surface: 3.5% and 5.0%, Upper surface: 3.5% and 5.0%, and Leading Edge
SD 7003 (8.5% thick)	Lower surface: 1.0% and 5.0%, Upper surface: 0.5% and 5.0% and Leading Edge
NACA 23012 (12% thick)	Lower surface: 1.0% and 4.5%, Upper surface: 0.8% and 3.5% and Leading Edge

The airfoils analyzed using the CFD data are relatively thin with thickness ratio, t/c , less than

Figure 4.1: Stagnation-point location, pressure-peak location, and pressure-peak value are compared from the LEFS algorithm (red) are compared with CFD results (blue) for the NACA 0012 at $Re = 3$ million. The dashed vertical line represents the stall angle of attack.

15% of the chord. Therefore, the direct approach is employed for the aerodynamic parameter estimation which does not require any pre-determined look-up tables.

Figure 4.1 shows the stagnation-point location, pressure-peak location, and the pressure-peak value estimated using the LEFS algorithm and compared with the actual values obtained from CFD. While certain inaccuracies are seen in the pressure-peak location, the value of the peak-suction pressure and the stagnation-point location compare well with the actual data even beyond stall. The values calculated from CFD require a dense pressure distribution near the leading edge to calculate the stagnation-point and the suction-peak pressure. The same is achieved via LEFS using just five pressure values, which demonstrates significant potential of the LEFS technique. Figure 4.2 compares the aerodynamic parameters determined directly from LEFS with the values from CFD. The velocity prediction matches excellently with the prescribed velocity used for the CFD computations. The angle of attack is determined accurately until stall and the force coefficients (C_l and C_s) compare remarkably well with the CFD data even beyond stall. The aerodynamic model in LEFS uses the equations of a parabola with leading-edge radius determined to be 0.0136 for NACA 0012 airfoil. Similar results are observed for the cambered SD7003 airfoil shown in Figure 4.3.

NACA 0012

Figure 4.2: Comparison of freestream velocity, angle of attack, lift coefficient, and suction coefficient estimated from LEFS (red) and the actual values from CFD (blue) for NACA 0012 at $Re = 3$ million.

SD7003

Figure 4.3: Comparison of freestream velocity, angle of attack, lift coefficient, and suction coefficient estimated from LEFS (red) and the actual values from CFD (blue) for SD 7003 at $Re = 3$ million.

4.1.2 Experimental Test Cases

The LEFS algorithm was applied to steady-flow experimental data obtained from test campaigns conducted by National Renewable Energy Laboratory (NREL). The available data contains pressure distribution for different wind-turbine airfoils tested under steady-flow conditions in the subsonic wind tunnel at the Ohio State University Aeronautical and Astronautical Research Laboratory (OSU/AARL). The Reynolds numbers used in these studies for all the different airfoil cases were 0.75, 1, 1.25, and 1.5 million.

The pressure values were extracted from the available data at five locations in the vicinity of the leading edge. The LEFS algorithm was utilized to estimate the local inflow velocity, angle of attack, and the lift coefficient using the five pressure measurements extracted from the experimental data. These estimated airdata quantities were compared to the actual experimental values to evaluate the effectiveness of the LEFS algorithm. The results for two airfoils, S809 [33] and LS0417 [59], are presented in this section. The locations of the five pressure tap measurements used on these airfoil models are listed in Table 4.2.

Table 4.2: Pressure port locations used to extract data for LEFS inputs from experimental data.

Airfoil	Pressure locations (x/c)
S809 (21% thick)	Lower surface: 1.5% and 4.88%, Upper surface: 0.4% and 3.2% and Leading Edge
LS0417 (17% thick)	Lower surface: 1.9% and 4.5%, Upper surface: 1.6% and 3.0% and Leading Edge

The airfoils considered in the experimental test cases have greater thickness compared to the airfoil analyzed with LEFS using CFD data. The geometry comparison of airfoils LS0417 and S809 with NACA 0012 is shown in Figure 4.4. The limitations of the direct approach for aerodynamic parameter estimation discussed in Section 3.4 are highlighted in the experimental

Figure 4.4: Airfoil profile of NACA 0012, LS 0417, and S809.

test cases. Therefore, look-up table approach appears to be the appropriate method for thicker airfoils. In the results presented in Figures A.2 and 4.6, the two methods for the estimation of angle of attack and force coefficients are compared with the true data from experiments.

Figure A.2 shows the aerodynamic quantities for LS0417 determined from the LEFS algorithm with just five pressure measurements as the input for two Reynolds numbers. The velocity predictions match closely with the actual velocities within 2.05 % and 1.93% for Reynolds numbers of 0.75 and 1 million respectively. The angle of attack prediction by LEFS using the look-up table approach exhibits close correlation with the actual α values within the pre-stall regime with a mean absolute error of 1.04° and 0.97° for flow conditions pertaining to 0.75 and 1 million, respectively. However, the direct approach exhibits greater deviations with absolute mean error of 1.33° in angle of attack for the 0.75 million case. Similar behavior is observed in the estimation of lift coefficient, with the LUT approach in the LEFS performing better than the direct method.

Figure 4.6 shows the results for the S809 airfoil for Reynolds number of 0.75 and 1 million. The difference between the results of the LUT method and the direct method is more prominent for the S809 airfoil case, which has a thickness of 21%. Since the direct approach fails to account for the thickness effects, large overshoots in the lift coefficient are seen for all the Reynolds-

number cases. The LUT approach in LEFS performs well for the angle of attack prediction matching the actual α values with an absolute error of 0.8773° within the pre-stall range, $-14.2^\circ < \alpha < 15.3^\circ$, for 0.75 million case. In comparison, the direct approach starts showing greater departure well before the onset of stall. The velocity estimation, which is a direct output of the LEFS algorithm, matches excellently with the true values.

The results validate the effectiveness of the LEFS algorithm in determining the aerodynamic parameters for the pre-stall regime. As expected, the direct approach is not suitable for thicker airfoils, which makes the LUT approach a better alternative.

Figure 4.5: Comparison of velocity, angle of attack and lift coefficient from LEFS and wind tunnel, for $Re = 0.75$ (a) and 1 million (b) for LS0417 airfoil.

Figure 4.6: Comparison of velocity, angle of attack and lift coefficient from LEFS and wind tunnel, for $Re = 0.75$ (a) and 1 million (b) for S809 airfoil.

4.2 Steady Flow (Post-Stall)

Aircraft typically operate in the linear regime of aerodynamics, where the flow is mostly attached and the boundary layer is thin. The lift and the angle of attack are linearly related in this regime. As the angle of attack is increased, the adverse pressure gradient on the upper surface and the shear stress cause the flow to lose momentum and eventually lead to separation. Flow separation is characterized by significant loss in lift which can result in loss of control. Therefore, identification of stall onset is highly desirable to avoid aerodynamic stall. In addition, operating close to the linear regime at the maximum-lift angle of attack is important for optimal aircraft performance. In this investigation, the application of the LEFS algorithm is explored as a potential approach for detecting the onset of stall and testing its applicability in the post-stall regime.

4.2.1 Stall Sensing

The results presented in the previous section show that the aerodynamic parameters are determined reasonably accurately by LEFS using just five pressure measurements near the leading-edge of the airfoil. Monitoring the lift coefficient along with the angle of attack determined using LEFS during flight is one potential approach for stall detection. However, this requires tracking the time history of the deduced aerodynamic parameters. It is highly desirable to have a robust method that can estimate the aerodynamic state of the airfoil at any arbitrary angle of attack independent of any history of angle-of-attack information. Based solely on the pressure measurements used as inputs, the LEFS estimates lower values of angle of attack in the post-stall regime as shown in Figure 4.7 using the results of α prediction for NACA 0012 as an example. It is evident from the plot that the same deduced value of α_{LEFS} could be obtained for two actual aerodynamic states of the airfoil. Therefore, additional information is required in LEFS to distinguish between the pre-stall and post-stall behavior.

It is well known that the flow separation on the airfoil surface is closely related to aerodynamic stall and this behavior is frequently used in literature for developing low-order methods

Figure 4.7: Angle of attack estimate from LEFS (red) compared with the actual angle of attack from CFD (blue). The value of α_{LEFS} deviates from the linear trend near stall which results in the same deduced α_{LEFS} (green marker) being predicted at two different aerodynamic states.

to predict loss of lift and stall at high angles of attack. Due to this relationship, the separation point is a valuable piece of information in stall detection and avoidance. In the current aerodynamic model, suction peak directly relates to the LEFS outputs and the suction peak drops as the trailing-edge flow separation approaches the nose region. The same behavior is reflected as a reduction in A , depending on the location of the separation point. The stall behavior or flow separation is governed by important airfoil characteristics such as thickness, camber etc. Therefore, in certain situations, the progression of the separation point from the mid-chord to the leading edge is slow causing the suction peak at the leading edge to diverge from the anticipated reduction in lift due to stall. In such cases, the stall signature is not successfully captured by the parameter A from LEFS. Figure 4.8 shows the variation in A as the angle of attack is increased beyond stall for two airfoils, NACA 0012 and S809. It is observed from the results that, for the NACA 0012, a critical value of A can be identified that indicates the onset of stall. For the S809 airfoil, the onset of stall occurs at $\alpha = 15.3^\circ$ while the value of A drops only when the angle of attack is close to 17° , because of which the α_{stall} estimate based

Figure 4.8: Variation of A with angle of attack for NACA 0012 and S809.

on the critical value of A overshoots the actual α_{stall} . The results show that supplementary information regarding the location of trailing-edge flow separation is essential to make LEFS a robust technique.

The stagnation-point location is the second possible option for stall sensing and numerous conventional stall-warning devices rely on the movement of the stagnation point to a pre-determined location. LEFS is capable of accurately estimating the stagnation-point location using just the five pressure measurements. However, identifying the critical location of the stagnation point that marks the onset of stall is crucial and experimental tests are necessary to determine the critical location. Since the idea in the current research is to avoid the need for expensive experimental testing, a different approach is pursued.

The concept presented in current research relies on directly identifying the location of flow separation using sensors mounted on the airfoil surface near the mid chord. A characteristic flow feature associated with flow separation is the reversal of flow direction or change in direction of shear stress. Numerous sensors based on different transduction mechanisms can successfully capture the changes in the flow direction and effectively identify critical aerody-

Figure 4.9: Angle of attack estimate from LEFS (red) compared with the actual angle of attack from CFD (blue). The value of α_{LEFS} deviates from the linear trend near stall which results in the same α_{LEFS} (green marker) being predicted at two different aerodynamic states. The flow separation status at 50% chord can be tracked by sensors mounted at that location.

dynamic flow features like separation-point location, flow reattachment and stagnation point. Bio-inspired artificial micro-structures based on optical transduction mechanism were tested to identify flow separation. Chapter 5 outlines the possible technologies available, and one such approach based on artificial micro-structures was evaluated in greater detail with wind tunnel experiments conducted at NCSU. Artificial microstructures immersed in the boundary layer can effectively respond to the changes in the boundary layer, and hence they have the potential to identify critical aerodynamic flow features such as stagnation point, laminar separation and flow reattachment, and separation. For the purpose of the current chapter, the knowledge of separation-point reaching a particular location (typically 50% to 70% chord) on the airfoil surface is assumed to be known. This information serves as an effective switch to help the LEFS algorithm in distinguishing between the pre-stall and the post-stall aerodynamic state of the airfoil as shown in Fig. 4.9.

4.2.2 Post-Stall

The identification of stall onset was explored by considering different parameters from the LEFS output in the previous section. It was concluded that additional sensors near the mid-chord of the airfoil can be used for stall sensing for any type of airfoil. The next aspect of post-stall application is to modify the LEFS approach to account for post-stall behavior. Kirchhoff-Helmholtz equations formulate the lift as a function of angle of attack and flow separation location. Kirchhoff theory is widely used in literature in the development of semi-empirical methods that model post-stall behavior, and a good review of the theory is presented in Refs. [60, 61]. The lift coefficient, C_l , under separated flow condition is given by:

$$C_l \simeq 2\pi \left(\frac{1 + \sqrt{f}}{2} \right)^2 \alpha \quad (4.1)$$

where f denotes the separation-point location as a fraction of chord length. Similarly, the expression for the chordwise suction force can be approximated from Kirchhoff's solution as:

$$C_s \simeq 2\pi\alpha^2 \sqrt{f} \quad (4.2)$$

Here, 2π is the lift-curve slope for incompressible flow. However, in practical viscous flows, 2π can be replaced by the actual lift-curve slope for an airfoil that can be determined by low-order computations.

As seen from the results presented in Sec. 4.1, LEFS is able to estimate C_l and C_s reasonably well even for higher angles of attack beyond stall. If the flow separation point is known for a given aerodynamic state, then it can be plugged into Equations 4.1 or 4.2, for C_l or C_s respectively, to obtain a new estimate of angle of attack. In the current approach, A vs. f look-up tables were pre-determined using XFOIL code and integrated in the LEFS technique. The value of A , calculated from the actual CFD test data, was used to compute f by referring to the look-up tables. The result from this approach for NACA 0012 airfoil is shown in Figure 4.10. Although, the α predicted by LEFS post-stall approach correlates much better to the actual α as compared

Figure 4.10: Angle of attack estimate from LEFS (red) and LEFS modified for post stall (green) compared with the actual angle of attack from CFD (blue). Angle of attack estimate from LEFS modified for post stall using the actual separation-point location from CFD is also co-plotted (black) for comparison.

to α_{LEFS} (direct approach without post-stall correction), the new approach is still not capable of estimating α at higher angles of attack. This discrepancy is attributed to the inaccuracy in f vs. A look-up tables generated using low-order codes and can be tested by using the actual separation location data from CFD. Figure 4.10 also shows the α estimated by the modified LEFS approach (black curve) with the f vs. A data used from the actual CFD computations, results in a much better comparison with the actual α . The idea of extending the LEFS concept to estimate aerodynamic parameters in post-stall flow is still at an exploratory stage, and the results shown are intended to demonstrate the limits of the LEFS application in the current form. Further work is required to extend the idea with a more reliable approach to relate LEFS outputs with the actual flow-separation location. Additional test cases need to be analyzed to fully understand the applicability of the LEFS approach in post-stall flow regime.

4.3 Unsteady Aerodynamics

Unsteady flows on airfoils are often characterized by complex flow features that strongly influence the aerodynamic performance of the airfoils and results in large deviations in the forces and moments from steady flow behavior. A characteristic phenomenon associated with unsteady aerodynamics is the formation of the Leading-Edge Vortex (LEV). For an airfoil executing time-dependent motion where the angle of attack increases with time, the unsteadiness of the flow causes a reduction in the adverse pressure gradients, thereby resulting in a delayed onset of flow separation, with the delay being determined by the kinematics of the pitch rate. As the angle of attack further increases, the flow separates near leading edge and results in the formation of a leading-edge vortex over the surface of the airfoil [61]. This vortex traverses the airfoil chord and induces additional lift as long as it stays on the airfoil surface. However, the significant increment in lift is also accompanied by a large nose-down pitching moment. The flow progresses to a state of full separation after the vortex reaches the trailing edge and convects into the wake, thereby causing an abrupt degradation in lift. This process comprising of the sequences of events that develop over the surface of the airfoil due to the unsteady motion at high pitch rates is referred to as dynamic stall [61, 62]. In engineering applications such as rotorcraft and wind turbines, large deviations in the loads and intense vibrations caused by this phenomenon of dynamic stall can be detrimental to the structural integrity of the rotor blades.

Numerous research efforts have been concentrated on understanding unsteady aerodynamics, with theoretical models by Theodorsen [63] and Wagner [64], and classical theories by Garrick [65, 66] dating back to the 1920s and 1930s that provided invaluable insight into the complicated physics of unsteady flows. Dynamic stall problem in helicopters, further thrust the research in this direction. Extensive experimental studies by McCroskey et al. [67], McAlister et al. [68], and Carr et al [69] helped significantly in understanding and predicting dynamic stall. The advancement of numerical schemes and computational capabilities enabled the detailed analysis of the complicated flow field around airfoils in unsteady simulations [70, 71, 72, 73]. Detailed analysis of the unsteady flow phenomena conducted by researchers has greatly con-

tributed in studying and finding methods to suppress dynamic stall. The understandings gained from theoretical, experimental, and computational investigations laid the groundwork for the development of semi-empirical methods for application in the initial stage of rotor design and analysis [74, 75, 76, 77, 78, 79].

LEV formation associated with unsteady aerodynamics is prevalent in various engineering applications. Some application areas such as helicopter aerodynamics [80] and wind turbines [81], are directed towards minimizing the damaging effects caused due to dynamic stall, while others, inspired by biological systems in nature, aim at exploiting the enhanced lift accompanying LEV formation to improve the performance of flapping wing MAVs for aerial reconnaissance inside buildings and confined spaces [82]. Furthermore, extracting energy from flowing fluids using oscillating or flapping wing provides another useful application of unsteady aerodynamic behavior in energy harvesting devices [83, 84, 85].

In all the practical applications, monitoring the sequence of events involved in LEV shedding, i.e., initiation, detachment, and termination, in real time can provide the opportunity to efficiently take advantage of the unsteady effects. Different types of experimental techniques offer the ability to successfully probe the unsteady phenomena, but employing the same techniques in real flight applications is not always feasible. Signatures of vortex formation on the airfoil surface can be captured in the surface pressure measurements. This provides a potential application for testing the capability of the LEFS algorithm. Typically, dense pressure information is necessary to identify the pressure gradients or stagnation-point location that can be associated with the LEV signatures. The objective of the current investigation was to explore the possibility of using just five pressure measurements near the leading-edge region via the LEFS technique to identify the events associated with unsteady flow behavior.

4.3.1 Background and Formulation

Research efforts at NC State directed towards unsteady aerodynamics lead to the development of a low-order method for modeling flows past airfoils undergoing arbitrary combinations of

pitch, plunge and surge motions [86, 87, 88, 89]. The primary focus was to model the formation and effects of leading-edge vortices. This method was built on the time-stepping approach given by Katz and Plotkin [90], in which a discrete vortex is shed from the trailing edge at every time step to satisfy the Kutta condition. Analogous to the classical thin-airfoil theory, the vorticity distribution over the airfoil, $\gamma(x)$, is taken as a Fourier series:

$$\gamma(\theta, t) = 2U \left[A_0(t) \frac{1 + \cos \theta}{\sin \theta} + \sum_{n=1}^{\infty} A_n(t) \sin(n\theta) \right] \quad (4.3)$$

where θ is a variable of transformation related to the chordwise x -coordinate as,

$$x = \frac{c}{2}(1 - \cos \theta) \quad (4.4)$$

and $A_0(t), A_1(t), \dots, A_n(t)$ are the time-dependent Fourier coefficients, c is the airfoil chord, and U is the x -direction component of the freestream velocity. The Kutta condition (zero vorticity at the trailing-edge) is enforced implicitly through the form of the Fourier series. The Fourier coefficients are determined as a function of the instantaneous local downwash (due to all bound and shed vorticity) on the airfoil by enforcing the boundary condition that the flow must remain tangential to the airfoil surface.

The flow parameters at the leading edge of the airfoil play a crucial role in the onset of leading-edge vortex formation. One of the most significant contributors is leading-edge suction force. The pressures near the nose region of the airfoil integrate to develop a net force. This force is actually a consequence of the stagnation point moving away from the leading edge when the airfoil is at an angle of attack. The flow is forced to travel back and curve around the leading edge. The curvature of the flow gives rise to a low-pressure region near the leading edge resulting in a net suction force. If the airfoil thickness decreases to zero, the leading edge radius also approaches zero resulting in the flow to turn exactly 180° at the leading edge. Figure 4.11 illustrates this scenario. The suction force can be expressed as a non-dimensional quantity

Figure 4.11: Flow around an airfoil with zero thickness [88].

given as:

$$C_s = \frac{|\vec{F}_s|}{q_\infty c} \quad (4.5)$$

where q_∞ is the freestream dynamic pressure and c is the airfoil chord. The suction coefficient can be related to A_0 term in the Fourier series for vorticity distribution as [91]:

$$C_s = 2\pi A_0^2 \quad (4.6)$$

Ramesh et al. defined a Leading-Edge Suction Parameter (LESP) as a criterion to predict the initiation of the LEV formation in unsteady flows. The LESP is a measure of the suction peak and can be determined at every time step of the unsteady thin-airfoil theory. Since A_0 can be expressed in a direct relationship with C_s , the LESP parameter was simply identified as:

$$LESP(t) = A_0(t) \quad (4.7)$$

This is an interesting and significantly useful condition formulated by Ramesh et al. [88]. They hypothesized the existence of a critical value of LESP for a given airfoil and Reynolds number which serves as a ‘start condition’ for the onset of LEV formation, independent of motion kinematics. The instantaneous LESP value exceeding the critical value indicates that LEV shedding is active. The critical value is empirically pre-determined for the given airfoil and Reynolds number using data from experiment or CFD for a single prototypical motion.

For a given airfoil geometry, A_0 is linearly related to the angle of attack in stationary flow.

$$A_0 = \alpha - \frac{1}{\pi} \int_0^\pi \frac{df_c}{dx}(\theta) d\theta \quad (4.8)$$

where the integral is based on the camberline of the airfoil, f_c . The formulation for asymptotic matching presented in Chapter 3 resulted in a direct relationship between the angle of attack and the LEFS parameters a' and A . Hence, in steady flow, the following connection can be established:

$$A_0 = \underbrace{\alpha}_{a'A} - \frac{1}{\pi} \int_0^\pi \frac{df_c}{dx}(\theta) d\theta \quad (4.9)$$

The aerodynamic behavior observed in viscous unsteady flows is often attributed to one or more time lags in the aerodynamics, primarily the boundary-layer convection lag. Simply stated, in the time that the boundary layer takes to progress from the leading edge to the trailing edge, the airfoil already pitches up to a higher angle of attack. Thus the pressure gradient that the boundary layer experiences is not the instantaneous pressure gradient. Narisapur et al. [58] showed that the LESP or A_0 value accounts for all the unsteady lag effects induced due to the motion kinematics. Analogous to this, in the LEFS formulation, the parameters a' and A are solved from an aerodynamic model that uses real pressure measurements. Therefore, a' and A are successfully able to reflect the unsteady effects that are captured by surface pressures and the time lag effects due to the motion kinematics are inherently part of the LEFS output. Hence, for unsteady aerodynamics the relationship between LESP, a' , A and angle of attack can be written as:

$$LESP(t) = A_0(t) = \underbrace{\alpha(t) + \phi_{motion}}_{a'(t)A(t)} - \frac{1}{\pi} \int_0^\pi \frac{df_c}{dx}(\theta) d\theta \quad (4.10)$$

Here ϕ_{motion} is used to represent the terms associated with the unsteady time lag effects due to motion kinematics. The accuracy of this correlation between LESP and a' and A is the key for the successful application of the LEFS algorithm for sensing critical LEV events.

4.3.2 Test Cases and Results

The objective of this investigation is to explore the possibility of correlating the surface-pressure signatures to the events associated with leading-edge vortices, such as initiation, pinch-off, and termination. The wealth of CFD data available from on-going and previous research studies [58] on unsteady aerodynamics at NC State was tapped for evaluating the effectiveness of the LEFS algorithm in identifying the critical LEV events, as depicted in Fig. 4.12.

Figure 4.12: Vorticity contours from CFD simulations to illustrate the critical LEV events.

NCSU’s REACTMB-INS code was used for CFD calculations, and the details of the computations and validation studies are discussed in Ref. [58]. Time variations of five pressure measurements were extracted from the simulations and post-processed through the LEFS algorithm to obtain time variations of a' and A . The LESP parameter (A_0) was estimated using A and a' , and compared with the actual LESP values (extracted directly from CFD) for the respective test cases. Three simulation cases were selected from the CFD database for the NACA 0012 airfoil at a Reynolds number of 3 million. The unsteady motions are pure pitching motions about a pivot location and the flow behavior over the airfoil is governed by a non-dimensional pitch rate, K . The specifications of the test cases, along with the associated critical unsteady flow features, are tabulated in Table 4.3.

Table 4.3: Unsteady flow simulation test cases for NACA 0012 airfoil at $Re = 3$ million.

Case	α (Deg)	K	Pivot (x/c)	Flow features
1	0-45-0	0.025	0.25	LEV initiation, pinch-off. and termination
2	0-35-0	0.005	0	No LEV formation
3	0-45	0.05	0.25	LEV formation with periodic shedding via multiple pinch-offs

Case 1

Figures 4.14 and 4.15 show the results of the LEFS algorithm applied to Case 1. In this case, as the airfoil undergoes pitch-up motion from $\alpha = 0$ with a pitch-rate of 0.025, the flow on the upper surface remains attached even as the angle of attack exceeds the steady stall angle. As the airfoil continues with the motion, the suction in the leading-edge region coupled with the adverse pressure gradient govern the LEV formation and the flow-field exhibits all the events associated with LEV formation, i.e., initiation, detachment, and termination. Figure 4.13(a) shows the variation of the actual LESP value throughout the motion co-plotted with the LESP value calculated from the LEFS algorithm, while Figure 4.13(b) plots the time variation of the parameter A from the LEFS output. The markers in both subfigures correspond to the time instant for which the vorticity and velocity contours are shown in subfigures (c) and (d) respectively. Observations show that the onset of the LEV formation happens when the LESP value exceeds a pre-determined value, as hypothesized by Kiran et al. This condition is defined as the starting criterion for the LEV initiation, and as soon as this condition is satisfied, the LEV shedding is considered to be ‘ON’. Additionally, the value of the parameter A also drops below a critical value, A_{crit} , which corresponds to the value of A at steady stall for NACA 0012 at $Re = 3$ million.

While the LEV shedding is ‘ON’, an oscillating pattern in the LESP variation is recognized at the time instants when the detachment of the shear layer takes place. Figure 4.14 shows that at the instant when the vortex pinches-off, a minimum in the LESP is reached. Hence, this condition can be used as the criteria for the identification of the LEV detachment event.

Figure 4.13: Onset of LEV formation for case 1 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant when the LEV formation initiates, for which the vorticity and velocity contours are shown in (c) and (d).

Figure 4.14: LEV detachment for case 1 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant when the LEV detachment occurs, for which the vorticity and velocity contours are shown in (c) and (d).

Figure 4.15: Termination of LEV shedding for case 1 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant when the LEV shedding terminates, for which the vorticity and velocity contours are shown in (c) and (d).

Finally, the time instant corresponding to the termination of the LEV shedding is shown in Fig. 4.15. At this instant, the value of the LEFS parameter A exceeds the defined critical value. This behavior can be attributed to the fact that the flow reattaches on the upper surface, and since the value of A is sensitive to the separation-point location, the flow reattachment is reflected by an increase in the value of A past the critical value. This observation leads to the hypothesis that the rise of A beyond the critical value can be used as an effective signature to signify the termination of LEV shedding.

Case 2

In this case, a similar pitching motion is followed as in Case 1 but with a low K value and with the pivot point located at the leading edge. At a high angle of attack, the boundary layer separates due to the adverse pressure gradients but the suction force is not sufficient to cause LEV formation. Therefore, the flow exhibits only trailing-edge separation without being accompanied by LEV formation. The flow field behavior can be effectively observed from the LESP variation shown in Figure 4.16(a). The start condition is never satisfied because the LESP fails to exceed the critical value.

Case 3

This case illustrates a scenario where the LEV formation is initiated but never terminated, resulting in the continuous formation and detachment of vortices from the leading-edge. The airfoil is pitched up to $\alpha = 35^\circ$ with $K = 0.05$ and held at this angle of attack for the rest of the unsteady simulation. Figure 4.17 shows the time instant when the LESP exceeds the critical value, thereby initiating the onset of LEV formation. Subsequently, the LESP exhibits an oscillating trend with multiple local extrema throughout the remaining motion. The vorticity contour for a time instant conforming to the LEV detachment shown in Figure 4.18(c) corresponds with the minimum attained in the LESP at the same time instant. This observation agrees well with the signature for LEV detachment. Furthermore, the start condition is

Figure 4.16: Trailing-edge stall for case 2 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant showing trailing-edge separation, for which the vorticity and velocity contours are shown in (c) and (d).

accompanied by the drop in A , however this value of A never exceeds A_{crit} for the remainder of the motion. Therefore, the condition for LEV termination is not satisfied, which agrees with the observation of continuing LEV shedding for the remaining part of the simulation.

In all the test cases considered in this investigation, the LESP prediction from the LEFS parameters, a' and A , matches excellently with the actual LESP values. It is worth noting here that the LEFS calculations are performed with only five pressure measurements near the leading-edge, while the LESP values calculated from the CFD simulations require a very fine distribution of pressures on the upper and the lower surface of the airfoil within a predefined distance from the leading edge. The integration of the pressure forces on discrete panels was required for the calculation of the net chordwise force component which is defined as the suction force. The non-dimensional suction coefficient was ultimately converted to the LESP parameter using Eq. 4.10. These results demonstrate the capability of the LEFS technique to identify the critical events associated with LEV formation in real time. The conditions for onset of LEV formation, detachment, and termination observed from the current results can be summarized as follows:

- LEV initiation : When LESP reaches a pre-determined critical value, LEV shedding starts. The value of A drops below the pre-determined critical (static-stall) value at this time.
- LEV detachment : When LEV shedding is on, a minimum in the LESP corresponds to pinch-off of the shear layer.
- LEV termination : When LEV shedding is on and A exceeds the critical value, LEV shedding stops.

Figure 4.17: Onset of LEV formation for case 3 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant when the LEV formation initiates, for which the vorticity and velocity contours are shown in (c) and (d).

Figure 4.18: LEV detachment for case 3 (NACA 0012). Comparison of LESP parameter from CFD and LEFS for the entire motion is shown in subplot (a). The quantity A from LEFS is shown in subplot (b). The marker represents the time instant when the LEV detachment occurs, for which the vorticity and velocity contours are shown in (c) and (d).

4.4 NREL S809 Rotating Blade Experiments

The Unsteady Aerodynamics Experiment (UAE) was conducted by National Renewable Energy Laboratory (NREL) with the primary objective of quantifying the full-scale, three dimensional aerodynamic behavior of horizontal-axis wind turbines [92]. A 10-meter diameter research wind turbine was tested in the world's largest wind tunnel located at the NASA Ames Research Center in Moffett Field, California. The tunnel test section is 24.4 m x 36.6 m (80 ft x 120 ft). The wind turbine was extensively instrumented to accurately measure several aerodynamic quantities for numerous test sequences. Figure 4.19 shows the two bladed turbine in the wind-tunnel test section.

Figure 4.19: The research turbine with instrumented blade mounted in the wind-tunnel test section.

The wind turbine blade used in the NREL test campaign was a two-bladed turbine with

Figure 4.20: Pressure taps and probe locations on the wind turbine blade used in the NREL UAE experimental campaign [92].

tapered and twisted blades that use the S809 airfoil from the root to the tip. Blade surface pressures, angle of attack and local dynamic pressure were measured at five radial locations on one blade. Five-hole probes with a spherical tip were used to provide the dynamic pressure, local flow angle, and spanwise flow angle measurements at a distance of 80% chord ahead of the blade. These probes were mounted at five radial locations: 34%, 51%, 67%, 84% and 91% span. The surface pressure distribution was captured with an array of 22 pressure taps instrumented at five primary radial stations: 30%, 47%, 63%, 80% and 95% span. Since the blade was tapered, the chord length at these locations were 0.711, 0.627, 0.542, 0.457, 0.381 meters, respectively. Figure 4.20 (Image taken from the NREL UAE report [92]) illustrates the pressure tap locations along the chord and the probe locations along the span of the S809 blade.

Similar to the steady-flow test cases, the pressure measurements from only 5 pressure taps

(out of 22) were acquired from the dataset and given as input into the LEFS algorithm. Although the same airfoil was employed in the blade section as used in the OSU airfoil model, the pressure tap locations were different in these tests. The selected pressure taps were located at 1% and 4% chord on the upper surface, 2% and 6% chord on the lower surface and on the leading edge of the S809 airfoil. The analysis using LEFS was performed at all the five radial stations and the look-up tables were developed by the same approach that was described in Section 3.4. The datasets from the test sequences in Ref. [92] provide variations of the normal force coefficient (C_n). To compare with the dataset listings, the lift coefficients in the look-up tables were transformed into normal force coefficients. The Reynolds number used in the look-up tables were 0.75, 1, 1.25 and 1.5 million. Three wind speed cases were studied in this research with the wind turbine blades rotating at a constant angular velocity of 70.38 rpm (7.37 rad/sec). The data files available from this campaign contains measurements after every 1.9 milliseconds for a total duration of 306 seconds. These values were averaged over each 1 second interval and used in the algorithm.

Since S809 airfoil was used for this test campaign, the LUT approach was adopted instead of the direct method for estimating the angle of attack and force coefficient. The local freestream velocity predicted by the LEFS algorithm at different radial locations on the turbine blade was compared with the actual inflow velocities at the blade sections, which were derived from the measurements of stagnation pressure available in the datasets. According to the NREL report, the pressure tap along the chord where the measured pressure attains a maximum was considered as the stagnation point, and the corresponding pressure at that location was used as the stagnation pressure. The resolution of the pressure taps near the leading edge (around 10 on the upper and lower surface) was deemed sufficient enough to extract the maximum positive pressure. The results for velocity estimations are summarized in Figures 4.21-4.22. These results demonstrate the remarkable capability of the LEFS algorithm to estimate the local inflow velocity using half the number of pressure taps used in the traditional stagnation-point method. Additionally, with the aid of XFOIL generated LUTs, the normal force coefficients

were predicted by the LEFS algorithm and they are co-plotted with the actual experimental values available from the tests.

The angle-of-attack range for the test sequences extended to very high angles beyond stall. The implementation of the algorithm was focussed only in the domain that spanned from -20 degrees to 20 degrees because the LUTs were generated for this range. However, since the velocities are determined as a direct output, the algorithm could be employed even at higher angles of attack for estimating the inflow velocities. Table 4.4 lists the mean absolute errors in local velocity prediction at the different span locations for two different freestream velocity cases and angle of attack less than 30 degrees. The LEFS algorithm also performs well for the C_n estimations at different span sections with greater deviations observed at the 30% span, which can be attributed to the prominent 3D flow effects in that region.

Table 4.4: Mean absolute error at different span locations for two freestream velocity cases.

Span (%)	6 m/s	10 m/s
30	0.3258	0.3490
47	0.2018	0.2361
63	0.2896	0.2549
80	0.5148	0.5459
95	0.1498	0.4943

The performance of the LEFS algorithm for a rotating blade demonstrates the algorithm's ability to sense the local flow velocity at different blade sections using just five pressure measurements.

(a) Span locations used.

(b) Estimated local flow velocities

(c) Comparison of normal force coefficient from LEFS (markers) and experiments (dashed lines) at different span locations

Figure 4.21: Comparison of the local flow velocities and normal force coefficient at different span locations for S809 rotating blade at a freestream velocity of 6 m/s.

(a) Span locations used.

(b) Estimated local flow velocities

(c) Comparison of normal force coefficient from LEFS (markers) and experiments (dashed lines) at different span locations

Figure 4.22: Comparison of the local flow velocities and normal force coefficient at different span locations for S809 rotating blade at a freestream velocity of 10 m/s.

Chapter 5

Separation-Point Detection

This chapter presents the micropillar-based approach explored for the detection of separation-point location. In this work, elastic microfence structures were tested under different flow conditions on a flat-plate setup. The sensor fabrication and transduction mechanism is discussed in Sec. 5.1. Details of the flat-plate model and the experimental setup is outlined in Sec. 5.2. The response of the microfences was recorded via optical microscope for flow over a flat plate at different freestream velocities. Experiments were also conducted to characterize the directional sensitivity of the microstructures by creating flow reversal at the sensor location to assess the sensor response. Section 5.4 shows the response of the microstructures under different flow conditions.

Artificial hair sensors for aerodynamic flow sensing have gained some interest from researchers over the past few years. Inspired by sensory structures in animals as diverse as bats [6], fishes [93], and insects [94], hair-cell arrays may enable real-time distributed surface-flow sensing providing rich data on flow over aircraft and other devices. The resulting technology may prove to be beneficial in systems like micro-aerial vehicles and wind turbines. It is believed that birds and insects exploit the sensing from various mechanoreceptors embedded in their wings to fine tune their flight performance [1, 2]. For instance, birds can sense the slightest changes in the airflow which enables them to stabilize in varying wind gusts and changing weather [3, 4]. Bats

have the remarkable ability to reverse flight directions at high speeds in short distances [5]. These outstanding capabilities are believed to be possible due to the fact that bats and birds can effectively interact with the surrounding airflow by sensing crucial information such as the flow speed, flow stagnation point, stall, and turbulence over the wing through numerous mechanoreceptors [6].

For flight systems, the knowledge of the incoming aerodynamic flow can be useful for stall sensing, envelope protection and control. The mechanical response of microstructures can potentially be used for surface-flow sensing, especially in MAVs where conventional sensors demand too much power and space and may not have the fidelity for aerodynamic control feedback. Artificial microstructures immersed in the boundary layer can effectively respond to the changes in the boundary layer, and hence have the potential to identify critical aerodynamic flow features such as stagnation point, laminar separation and flow reattachment, and separation.

Micropillars can also provide a means of quantifying shear stress if the height of the micropillar is restricted within the viscous sublayer. The velocity profile can be assumed to be linear within the viscous sublayer of the boundary layer and hence, the pillar tip deflection can be indirectly related to the wall shear stress [95]. An array of cylindrical micropillars could provide complete spatial and temporal resolution of the shear stress [96, 97, 98, 99], making it a convenient and inexpensive alternative to conventional indirect shear stress measuring techniques, such as hot wire/film, oil interferometry, laser doppler velocimetry, or particle image velocimetry [100].

Smart materials systems such as carbon nanotube arrays [101, 102, 103] and piezoelectric microstructures [104, 105] can directly convert pillar deflection to electrical output. More complicated sensor designs and transduction mechanisms involve cylindrical microstructures attached to flexible platforms, where the rotational moment of the base of the pillar displaces the membrane underneath resulting in an electrical response in the form of capacitance, inductance or resistance change [106, 107, 108]. The moments at the bases of 1-mm cylindrical micro-pillars have been modeled to assess their ability to detect the boundary layer changes [93, 109].

This research presents the experimental response of elastic surface topographies using optical measurement to track tip deflections. Most previous studies involving characterization of flow properties using elastic topographies are conducted for pipe/duct flows or wall jet flows. In this study, the response of elastic microfence structures mounted on a flat plate in a subsonic wind tunnel was recorded via optical microscopy through the sample. Image stabilization algorithms along with integrated stationary fiducial-mark topographies were utilized to compensate for wind tunnel vibrations.

5.1 Sensor Fabrication and Transduction

Elastic surface topographies, designed to be responsive to profile drag within the viscous sublayer, comprise the microfence structures. Templates (molds) were generated by patterning epoxy substrates using laser ablation, which enabled rapid design changes. Silicone materials of various elastic moduli were cast on the templates to generate the sensor surface. The technique allowed for good control of the depths of the pattern features. The sensing elements consisted of microfences with an aspect ratio of at least three, and of various orientations relative to the freestream flow vector, to enable determination of shear stress in different directions relative to the airflow direction. Stationary features (fiducial marks, long lines or dots that should not move in the airflow) were provided to enable position referencing (Figure 5.1). Featureless regions were included to prevent disruption of the airflow from a sensing element upstream. The fabrication of the microfence structures was done by our collaborators at NASA Langley Research Center.

Since the intention was to minimize disruption of the airflow by keeping the structures within the viscous sublayer, feature heights on the order of 50-100 μm were targeted. These dimension requirements defined the resolution requirements for an optical visualization technique. An optical microscope was constructed with emphasis placed on minimizing the “footprint”, i.e., the X-Y dimensions, in an effort to reduce interactions of the device with the airflow as well as to enable ready implementation into the designed wind-tunnel model. The resultant instrument

Figure 5.1: Microfence structures ((a) top view, (b) isometric view, (c) isometric view shown with higher magnification, (d) single microfence structure and four fiducial marks).

Figure 5.2: Schematic (A) and photograph (B) of the shear stress-sensor-prototype microscope for optical tracking.

was approximately 8 in $\times$ 8 in $\times$ 10 in (length $\times$ width $\times$ height) and consisted of a 10 $\times$ microscope objective with a long working distance connected to an in-line light source and a charge-coupled device (CCD) camera. The camera used was a progressive-scan CCD with a resolution of 1.3 Mp and frame rate of 15 fps. Integration into a wind-tunnel model also guided the design of the microscope arrangement (Figure 5.2). Because the current work aims

at detecting surface-flow features rather than accurate shear-stress measurement, the decision was made to use an inexpensive camera rather than one with a much higher frame rate such as those used in earlier works [96, 97, 98, 99]. Success with such a lower-cost camera would allow for the research to have an easier path for transitioning the technology out of the laboratory and into potential real-world applications.

5.2 Experimental Setup

To evaluate the response of the elastic microfence structures, the sensing elements were installed on a flat plate model and subjected to controllable flow conditions in the subsonic wind-tunnel at North Carolina State University (NCSU). The details of the wind tunnel facility and flat-plate model are presented in the following subsections.

5.2.1 Wind-Tunnel Facility

The facility at the North Carolina State University is a closed-circuit subsonic wind tunnel with a 0.81-m high, 1.14-m wide, and 1.17-m long test section. The wind-tunnel fan is equipped with variable-pitch blades allowing the velocity in the test section to be continuously varied up to a maximum of 40 m/s. The turbulence levels have been determined to be less than 0.33%. Upstream of the test section and forward of the contraction section is a settling chamber consisting of an aluminum honeycomb screen followed by two stainless steel anti-turbulence screens. A breather is located downstream of the test section to ventilate the tunnel to room pressure.

5.2.2 Flat-Plate Model

A (0.66-m chord) flat-plate model with an elliptical leading edge was fabricated so that the shear sensors could be installed and tested in the middle portion of the upper surface of the plate. The model was constructed with surface pressure taps in the leading edge and on the upper surface of the flat plate. The model also has a trailing edge flap that can be adjusted to control

Figure 5.3: Flat plate model in the wind tunnel ((a) drawing in which the fairing is not shown for clarity, (b) view of the model in test section, (c) view of the fairing under the test section housing the camera and microscope assembly).

the stagnation point at the leading edge of the model. This capability provides better control over the leading-edge suction peak. Figure 5.3 shows the flat-plate setup inside the wind-tunnel test section. The microfence sensors were installed on a small quartz disk approximately 3.175 cm in diameter, which in turn was mounted at around 50% of the chord. A faired support structure was designed to house the camera assembly, and provide internal access for installing the prototype instrument below the quartz plate.

5.3 Data Processing

The response of the elastic microfence structures was measured using optical microscopy. Section 5.1 provided the details of the camera setup and assembly. This section presents the post-processing methodology adopted to transform the raw videos recorded during experiments into useful information.

Wind tunnel facilities are prone to vibrations. For the experimental setup used, the vibrations result in relative motion between the camera setup and microfence structures which makes the measurement of microfence tip deflection a challenging task. Although measures were taken in designing the current experimental setup to minimize any mechanical vibrations, flow-induced vibrations were still present during the experimentation. Hence, the videos recorded during the testing required post-processing to mitigate the effects of vibrations. Image-processing algorithms from MATLAB were employed for image stabilization.

The images obtained from the recorded videos were first converted to grayscale format such that every pixel location has an associated intensity value. These intensity values were adjusted to enhance the contrast between the substrate background and the microfences by mapping to new values such that 1% of data is saturated at low and high intensities of the original grayscale image. Figure 5.4 compares the raw image (left) with the enhanced version (right). Intensity-based image registration technique was used to transform the video frames and correct the relative motion induced in the video due to vibration. This approach generates a similarity metric between a reference image and the moved image for spatial referencing. The similarity metric is maximized iteratively based on an optimizing function to create a transformation matrix that is used to realign the moved image with the reference. In the wind-tunnel experiments, the first frame of the video (when the freestream velocity is zero) was taken as the reference image for the test cases and all the subsequent frames were processed and realigned based on the initial frame. This process of transformation mitigated the relative motion induced by vibration to a great extent and facilitated the tracking of the microfence tip deflection.

In the final post-processing step, a region of interest was manually defined around the tip of the microstructures and the intensity values were monitored at each pixel in that region. Since the intensity values at the microfence tip region are affected by deflection, tracking the pixel intensities above a set cutoff value proved to be a successful method for determining the deflection of the microstructures. The movement of the tracked tip area was measured in pixels

Figure 5.4: Raw image (left) from the video recorded in the experiments compared to the enhanced version (right) with better contrast after intensity adjustment.

and this was converted to micrometer values based on the known length of the microfences.

5.4 Results

Microfence tip deflections in the flat-plate turbulent boundary layer were observed in the wind tunnel for different freestream velocities. Two different experimental setups were designed to evaluate the performance of the micro-sensors. The first test case was aimed at investigating the microfence response under the influence of different freestream velocities. The flat plate model shown in Figure 5.3 was used to mount the microfence structures. The second experimental case was designed to assess the directional sensitivity of the microfences by subjecting it to local flow reversal. The results from the two test cases are presented in the following subsections.

5.4.1 Case 1: Effect of freestream velocity

In this test case, the microfence tip deflection was measured under different freestream velocities. The elastic microstructures were installed at a location of 50% chord on the flat plate setup shown in Figure 5.3 with a chord length of 26 inches and subjected to four freestream velocities of 15.2, 21.6, 24.9, and 30.5 m/s, corresponding to local Re_x of 3.4×10^5 , 4.8×10^5 , 5.9×10^5 , and 6.8×10^5 , respectively. For each Reynolds-number case, the sensor response was recorded

as the velocity was steadily ramped up from zero to the required velocity. The first image in the frame sequence was treated as the reference image for vibration stabilization and the tip of the microfence was identified by using the maximum intensity values of the pixels in the grayscale images in a region of interest (covering the microfence under investigation). The images in Figures 5.5(a) and 5.6(a) compare the starting reference frame to a frame at the steady freestream velocities of 24.9 m/s and 30.5 m/s.

(a) Reference frame (left) and frame with deflection (right).

(b) Microfence tip regions marked based on intensity values for reference image (left) and deflection image (right).

(c) Initial and final tip regions shown overlapped with circular dots indicating the respective centroids.

Figure 5.5: Microfence tip deflection for U_∞ of 24.9 m/s and Re_x of 5.9×10^5 . Flow is from right to left.

(a) Reference frame (left) and frame with deflection (right).

(b) Microfence tip regions marked based on intensity values for reference image (left) and deflection image (right).

(c) Initial and final tip regions shown overlapped with circular dots indicating the respective centroids.

Figure 5.6: Microfence tip deflection for U_∞ of 30.5 m/s and Re_x of 6.8×10^5 . Flow is from right to left.

The microfence tip areas based on the pixel intensity levels are shown for two velocity cases in Figures 5.5(b) and 5.6(b). Since the intensity values near the tip are affected by the deflection, the tracked tip area varies, so the movement of the centroid of the tip was used to give an estimate of the pillar tip deflection, as illustrated in Figures 5.5(c) and 5.6(c). The length of the base of the microfence is approximately $93 \mu m$ and this information is used to translate the pixel deflections to micron scale. The estimated deflections for all the velocity cases are tabulated in Table 5.1 and plotted in Figure 5.7, along with the corresponding values in micrometers, μm .

Table 5.1: Microfence deflections observed for different freestream velocity cases.

U_∞ (m/s)	Δx (px)	$\Delta x(\mu m)$
0	0	0
15.28	2.67	1.05
21.61	4.25	1.67
26.47	7.45	2.93
30.56	14.27	5.61

Figure 5.7: Variation of microfence tip deflections with freestream velocity.

5.4.2 Case 2: Detection of reversed flow

The second test case was designed to characterize the response of the microfence structures under reversed flow conditions. Reversed flow on the surface of the plate at the location of the microfences was achieved by using a wedge-shaped obstruction placed in front of the microfences. The shape and location of the wedge was selected by using surface oil-visualization to ensure that the surface flow was reversed at the location of the microfence structures. A movable 3D-printed wedge shaped body (having a 30 degrees wedge angle, as shown in Figure 5.8) was fabricated for this purpose. A linear actuator was used for moving the wedge. In the default unactuated position, the wedge is located so that it is not in front of the sensor. Upon actuation, it slides in front of the microfences mounted at the mid-span location. Figure 5.8 shows the flat plate inside the wind tunnel with the wedge installed on it. The test sequence with the corresponding micropillar response at critical flow conditions is presented in Figure 5.9 for a U_∞ of 24.9 m/s.

Figure 5.8: The 3D printed wedge with 30 degrees angle used to create flow reversal (left). A modified flat-plate setup with the wedge mechanism installed on the top surface, shown here in the unactuated position (right).

(a) State - 0 : Zero freestream velocity and wedge placed away from microfence location.

(b) State - 1 : Desired freestream velocity is attained but the wedge is still away from microfence location.

(c) State - 2 : Desired freestream velocity and wedge is moved in front of the microfence location.

(d) State - 3 : Desired freestream velocity and wedge is moved away from the microfence location.

Figure 5.9: Micropillar image (micrograph) with a red vertical line marking the initial reference location of the microfence tip. Top view of the flat plate (right) shows the corresponding location of the wedge. The blue markers indicate the direction of the freestream flow and the white arrows indicate the local flow direction observed by the microfence sensor.

The effect of reversed flow on the microfence tip deflections for two freestream velocities is presented in Figure 5.10. An increase in the deflection can be observed as the velocity is steadily increased from zero to the desired value and the wedge is positioned away from the microfence location. When the wedge is actuated towards the sensor, a decrease in deflection values can be observed which eventually become steady as negative deflection values because the flow is reversed at the sensor location in the presence of the wedge. Finally, as the wedge is moved back, the microfence tip again deflects in the freestream flow direction as evident from the plots in Figure 5.10. Although the data still exhibits fluctuations due to the flow-induced vibrations, a noticeable trend can be easily detected that demonstrates the directional response of the microfence pillar. It is noted that the scatter in the deflection of the microfence pillar in the current work is much larger than that reported in [110, 99]. It is believed that the increased scatter in the current work is primarily due to the use of a lower-cost camera with a lower frame rate and also due to the mounting of the camera as part of the same structure as the airfoil holding the sensor array. For the purpose of detecting surface-flow reversal, however, the results show that the scatter does not pose a major problem. It is worth noting that the importance of the current research lies in the scale of the wind-tunnel investigations conducted for the characterization of the microfence response to flow conditions on an airfoil surface.

Figure 5.10: The deflection (in pixels) of the microfence sensor with time as the velocity is increased and the wedge is translated towards the sensor location and then moved away.

Chapter 6

Conclusions and Future Work

Real-time measurement of aerodynamic parameters during aircraft flight is a capability that will be greatly beneficial for load monitoring and flight performance enhancement. Aerodynamic state refers to the magnitude of the incoming flow velocity, U_∞ , effective angle of attack, α , and the sectional force coefficients, C_l and C_s . The knowledge of the sectional aerodynamic state of various wing and tail sections could be used for flight control and envelope protection by effectively adapting the wing geometry using cruise flap or other geometry changes. Dangerous and unrecoverable stall situations can be anticipated and successfully avoided by possessing accurate knowledge of the complex flow environment. Flow-field sensing on the horizontal tail from a few surface pressure measurements can be used to detect the loss of effectiveness of the tail due to impingement of the wing wake and allow for real-time estimates of the control-surface performance. Apart from aircraft applications, the knowledge of the aerodynamic operating condition could be advantageous in horizontal-axis wind turbines, in which peak power generation requires operation of the blades close to aerodynamic stall.

Most airfoils designed for subsonic flow have a non-zero leading-edge radius, which makes it possible to approximate the leading-edge region with a parabolic profile. Many researchers have taken advantage of simplified flow solutions to better understand the characteristics of complex flow phenomena. Current research employed the same fundamental approach but was focused

towards the application of flow sensing. A novel leading-edge flow sensing (LEFS) method was built using the equations of flow past a parabola that fits the pressures measured on the airfoil surface to the pressure distribution from the exact potential solution for the parabola. Pressures sensed at five discrete locations near the leading edge of an airfoil were given as input to the algorithm to solve the aerodynamic model using a simple nonlinear regression. The inflow velocity and the stagnation-point location were computed directly from LEFS, while the estimation of angle of attack and lift-coefficient from the LEFS outputs was explored using two different methods. A direct technique was formulated based on the asymptotic matching of flow solution in the parabola and the airfoil. The inner limit of thin-airfoil theory solution was matched with the outer limit of the potential solution for a parabola to derive a direct relationship between the LEFS outputs, a' and A , and the aerodynamic quantities, α and C_l . Pressure data near the leading-edge of airfoils was extracted from CFD simulation and wind-tunnel experiments to assess the effectiveness of the LEFS technique in steady pre-stall flow conditions. Results showed that the direct approach performed remarkably well for airfoils with thickness less than 15% of the chord. However for airfoils with thickness greater than 15% chord, the results showed that the direct approach exhibited inaccuracies in the α and C_l predictions. Therefore, a look-up table approach was proposed in which a low-order airfoil code was used to develop look-up tables for different airfoils. The results for the LEFS look-up table approach compared well with the actual data, irrespective of the type of airfoil. The LEFS algorithm was also applied to the pressure measurements obtained from the surface of a rotating wind turbine blade. The experiments were part of the UAE test campaign conducted by NREL. Discrete pressure measurements taken near the leading edge were used in this research to test the algorithm for a rotating blade where 3D flow aspects come into play. The results presented for the velocity estimation and the normal-force predictions agreed well with those observed in the experiments.

Further, the application of the LEFS technique was explored for stall sensing and for aerodynamic parameter estimation in steady flow post-stall regime. Stall is characterized by flow

separation on the airfoil surface which results in significant loss of lift. Identification of stall-onset is beneficial for preventing loss of control and dangerous stall situations. Flow separation is characterized by significant loss in lift which can result in loss of control. Therefore, identification of stall onset is imperative to avoid the dangerous stall situation. In addition, operation of aircraft close to the linear regime at the maximum-lift angle of attack is important for optimal aircraft performance. In this investigation, the application of the LEFS algorithm is assessed as a viable approach for detection stall onset. The results proved that additional information regarding the location of flow separation on the airfoil surface was necessary. Of several possible approaches, bio-inspired microfence sensing elements were employed to capture signatures of flow separation in the form of flow reversal.

Elastic surface topographies designed to be responsive to near-wall flow behavior were mounted on a flat-plate experimental setup for characterization under real flow conditions. The response of the sensors at different freestream velocities and directional sensitivity under reversed-flow conditions were evaluated during the wind-tunnel experiments. Image-processing algorithms were employed for image contrast improvement and for minimizing the error induced due to vibrations observed in the wind tunnel. It was observed that the microfence tip deflection increased with an increase in freestream flow velocity at an expected trend as the force acting on the microfences squared with the local flow velocity. Although vibrational fluctuations still existed in the measurement data, the response of the microfences under reversed-flow conditions adhered to the local flow behavior as observed in the results. In any wind-tunnel facility or even in practical applications, the vibrations are unavoidable which hinders the application of these microfences as accurate shear-stress sensing tools due to the measurement error involved in the current optical transduction technique. However, the results demonstrated the potential of microfences as a successful technique for identifying critical aerodynamic flow features such as stagnation point, separation point, laminar separation, and flow reattachment.

The methodology for post-stall application of LEFS was devised using Kirchhoff's theory that model the lift coefficient as a function of angle of attack and separation-point location.

The knowledge of separation location was assumed to be known for the initial results presented in this research. Although the new technique improved the results for α prediction near stall, additional work is required for the accurate implementation of flow separation in the LEFS aerodynamic model to encompass higher angles of attack beyond stall.

Finally, the application of LEFS in unsteady flow was investigated to detect signatures associated with LEV formation. Real-time knowledge of the unsteady flow phenomena holds significant potential for exploiting the enhanced-lift characteristics related to vortex formation and inhibiting the detrimental effects of dynamic stall in engineering applications such as helicopters, wind turbines, bio-inspired flight, and energy-harvesting devices. Computational data was used to assess the capability of the LEFS outputs to identify the signatures associated with vortex formation, i.e. onset of vortex shedding, detachment, and termination. The results showed distinct correlation between the LEFS outputs and the LEV signatures that could be used in real-time to identify the critical events associated with unsteady phenomena. Further work is required to test the approach for additional cases such as cambered airfoils undergoing unsteady motions.

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APPENDIX

Appendix A

Steady Results

Additional results of the LEFS algorithm applied to steady flow conditions are presented in this appendix. In all the investigations, five pressure measurements and their locations are extracted from the computational or experimental test data and this information is processed through the LEFS algorithm to determine the sectional aerodynamic parameters. The values of leading-edge radius, r , and the camberline slope, λ , used to generate the parabola for different airfoils are tabulated in Table A.1.

Table A.1: Parameters used for generating the parabolic profile for different airfoils.

Airfoil	r	λ
NACA0012	0.0135	0.000
NACA23012	0.014	0.0218
SD7003	0.0103	0.0093
LS0417	0.026	-0.0072
S809	0.0136	-0.0101

A.1 NACA23012

NACA 23012

Figure A.1: Comparison of freestream velocity, angle of attack, lift coefficient, and suction coefficient estimated from LEFS (red) and the actual values from CFD (blue) for NACA 23012 airfoil at $Re = 3$ million. The dashed vertical line represents the stall angle of attack.

A.2 LS 0417

Figure A.2: Estimated velocity from LEFS algorithm (markers) compared with the actual velocity (dotted lines) for LS0417 airfoil at different Reynolds numbers.

(a) $Re = 0.75$ million

(b) $Re = 1$ million

Figure A.3: Comparison of the deduced α and C_l from LEFS and wind tunnel, for $Re = 0.75$ (a) and 1 million (b) for LS0417 airfoil. The dashed vertical line represents the stall angle of attack.

(a) $Re = 1.25$ million

(b) $Re = 1.50$ million

Figure A.4: Comparison of the deduced α and C_l from LEFS and wind tunnel, for $Re = 1.25$ (a) and 1.5 million (b) for LS0417 airfoil. The dashed vertical line represents the stall angle of attack.

A.3 S809

Figure A.5: Estimated velocity from LEFS algorithm (markers) compared with the actual velocity (dotted lines) for S809 airfoil at different Reynolds numbers.

(a) $Re = 0.75$ million

(b) $Re = 1$ million

Figure A.6: Comparison of the deduced α and C_l from LEFS and wind tunnel, for $Re = 1.25$ (a) and 1.5 million (b) for S809 airfoil. The dashed vertical line represents the stall angle of attack.

(a) $Re = 1.25$ million

(b) $Re = 1.50$ million

Figure A.7: Comparison of the deduced α and C_l from LEFS and wind tunnel, for $Re = 1.25$ (a) and 1.5 million (b) for S809 airfoil. The dashed vertical line represents the stall angle of attack.